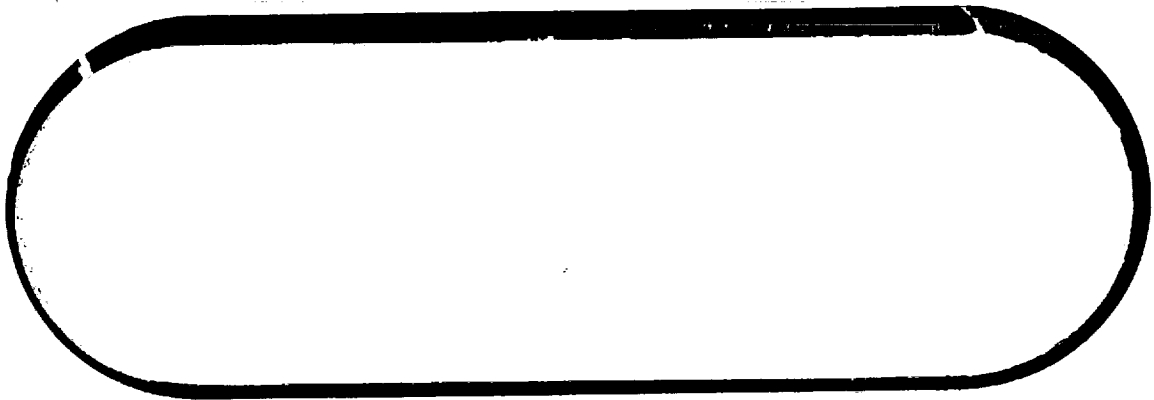


BOEING



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THE HELICOPTER SIZING AND PERFORMANCE
COMPUTER PROGRAM (Boeing Vertol Co.,
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USER'S MANUAL FOR HESCOMP THE HELICOPTER SIZING AND PERFORMANCE COMPUTER PROGRAM

Developed under
CONTRACT No. NAS 2-6107 (Study of the Methodology for Evaluation
of an Interurban and Intraurban V/STOL
Transportation System)

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FOREWORD

HESCOMP, the Helicopter Sizing and Performance Computer Program, provides helicopter designers with the same capability for sizing and performance calculations that VASCOMP II provides for fixed-wing aircraft designers.

Since in time the program will change to reflect new thinking and grow to include more sophisticated methods of simulating advanced helicopters, this User's Manual is loose-leaf bound to facilitate updating of the program.

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NOTE

Section 5.3 contains a definition of program input variables and indicators; section 6.2 lists the major diagnostic error printouts and describes their probable cause. For ease of reference, these sections are printed on blue and green paper, respectively.

1.0 INTRODUCTION

1.1 BACKGROUND

HESCOMP is a helicopter sizing and performance computer program very similar in format and operation to VASCOMP II, the V/STOL Aircraft Sizing and Performance Computer Program, described in Reference 1. This similarity is dictated by the requirement to obtain compatibility in both usage and results when using HESCOMP and VASCOMP II in helicopter-V/STOL aircraft comparative design studies. The program's purpose is to rapidly provide helicopter sizing and mission performance data. The program can be used to define design requirements, such as weight breakdown, required propulsive power, and physical dimensions of aircraft which are designed to meet specified mission requirements. It is also useful in sensitivity studies involving both design trade-offs and performance trade-offs.

During formulation of the program, the following guidelines have been followed:

1. The program should maintain generality and flexibility - A program of this type must be comprehensive and flexible in order to permit an accurate simulation of many types of helicopter configurations. It must be capable of approximating the design process involved in layout and sizing of a wide variety of helicopters and synthesizing the performance of these aircraft.
2. The program should be easy to use - In order to minimize hand computation of input data, the input to the program primarily consists of a series of single point values specifying, for example, main rotor disc loading, solidity, twist, aspect ratio, taper ratio, etc. of the wing, tail, and rotor pylons (where applicable), the geometry of the fuselage, the type of propulsion system, a description of the mission profile, and weights of fixed equipment, fixed useful load and payload. Where necessary to adequately describe certain functional relationships, the input is in tabular form. However, since preparation of data for tabular input is generally more cumbersome and time consuming, this form of input has been kept to a minimum.
3. The program should minimize computation time - In order to minimize computation time, the program makes ample use of optional computation paths. To eliminate large quantities of null arithmetic, it avoids calculations which do not apply to the particular aircraft being studied. This is accomplished by means of a series of input indicators that specify the calculations to be performed.

4. The program should be well balanced - The program should not be extremely sophisticated in one detail and yet extremely simple in another. To offset the possibility of this occurrence, great care has been taken to examine methods used to describe the helicopter and its operation.
5. The program should be compatible with VASCOMP II - In order to insure program compatibility, care has been exercised in planning the input/output format. The input sheets are similar (and in a few cases - identical) to those of VASCOMP II. The output format is the same except for the additions of those output quantities peculiar to helicopter performance. Further, this User's Manual is identical in format to the VASCOMP II User's Manual. In addition, HESCOMP utilizes (unchanged) the engine cycle library, propeller tables, and propeller short form performance method developed for VASCOMP II.

1.2 APPLICATION

The program has two primary independent applications and a third which is a combination of the first two. It may be used for the sizing of helicopters for which the type of aircraft and the mission profile are specified. Alternatively, it may be used for mission calculations for aircraft for which sizing details (gross weight, fuel available, engine power and fuel consumption, etc.) are known. As a combination of these two capabilities, the program may be used to first size a helicopter for a given mission and then calculate the off-design point performance for other missions. The option of calculation to be used is specified to the program by means of an input "option indicator."

The program has been written in a manner to make it directly applicable to sensitivity studies to determine the effect of variations in weight, drag, engine characteristics, etc. This is accomplished by use of incremental multiplicative and additive factors applied to the gross weight, component drag and fuel required equations. For the most part, the multiplicative factors are nominally equal to unity and the additive factors are nominally equal to zero. However, to determine the effect, for example, of a 10 percent increase in drive system weight, the appropriate multiplicative factor can be set to 1.10 and the sizing program rerun.

The program contains size trends equations which reflect the variation of helicopter dimensions with gross weight, detailed statistical weight trends equations, a routine for sizing of engines to match airframe requirements, a comprehensive library of engine cycle data, a library of rotor cycle data, and a variety of optional procedures for calculating rotor and propeller (cruise only) performance.

The program can be used to study any single or tandem rotor pure, winged, compound, or auxiliary propulsion helicopter (see Table 1-1).

TABLE 1-1. HELICOPTER CONFIGURATIONS WHICH MAY BE STUDIED USING HESCOMP.

Helicopter Type (Both Single & Tandem Rotor)	Additional Lift/Propulsion System Components Which Must be Added to "Pure" Conf.	Wing	Propeller for Auxiliary Propulsion	Auxiliary Independent Engines	Type of Auxiliary Independent Engines		
					T/Shaft	T/Fan	T/Jet
Pure Helicopter							
Winged Helicopter		X					
Compound Helicopter							
(1) Coupled (prim. engines drive auxiliary propulsion system)		X	X				
(2) Auxiliary independent propulsion system							
(a) T/Shaft engine		X	X	X	X	X	X
(b) T/Fan engine		X	X	X	X	X	X
(c) T/Jet engine		X	X	X	X	X	X
Auxiliary Propulsion Helicopter							
(1) Coupled (prim. engines drive auxiliary propulsion system)			X				
(2) Auxiliary independent propulsion system							
(a) T/Shaft engine			X	X	X	X	X
(b) T/Fan engine			X	X	X	X	X
(c) T/Jet engine			X	X	X	X	X

2.0 SPECIFICATION OF HELICOPTER CHARACTERISTICS

Specification of aircraft characteristics to the program is made in a variety of ways: through use of input indicators which specify the types of calculations to be made; through use of weights factors and constants; aerodynamics data; propulsion information; and mostly through use of nondimensional geometric information.

2.1 HELICOPTER GEOMETRY

It is assumed that a typical sizing analysis starts with known payload characteristics, both in terms of payload weight and volume requirements. The volume requirements are usually reflected in length, height, and width of the constant diameter (cabin) section of the aircraft. Adding a nose and tail section of reasonable fineness ratio onto the cabin sections would complete the fuselage geometry if this were an airplane (as sized by VASCOMP II). In a helicopter, however, additional geometric characteristics must be determined before the external fuselage dimensions are completely defined.

For example, in the case of the single rotor helicopter, the total fuselage length (in addition to the nose, tail, and constant diameter sections) includes the tail boom, the length of which, in turn, is established by the tail rotor diameter and the need to maintain a reasonable gap between the main and tail rotor discs. Additionally, the tail boom length itself is affected by the relative position of the main rotor on the fuselage. Vertical tail geometry is determined both by dimensional constraints and the need to fulfill directional stability requirements (e.g., sufficient vertical tail area to counteract main rotor torque in the event of tail rotor loss). So, although the basic cabin internal dimensions are fixed, the external overall dimensions can vary widely, depending on how conflicting requirements are resolved.

In the case of the tandem rotor helicopter, not even the internal cabin dimensions are necessarily constant. For example, the need to require a certain level of external configuration compactness (by specifying a high overlap/diameter ratio) can result in overall fuselage dimensions which directly conflict with internal volume requirements.

Wing geometry may be dictated by maneuver "g" requirements, a specified wing loading or aspect ratio, or even propeller tip/fuselage clearance (in the case of a compound helicopter with wing-mounted propellers).

Primary and auxiliary independent engine nacelle size is set

by the type of engine and its size (which, in turn, is dictated by power requirements).

Figures 2-1 and 2-2 of hybrid single and tandem rotor helicopter configurations illustrate the type of information concerning the helicopter geometry which may be required of the user. Tables 2-1 and 2-2 illustrate typical values of selected geometric characteristics for various aircraft. A complete list of input geometric variables is included in Section 5.3.1.

2.2 PROPULSION SYSTEM

This program permits the use of either a single, primary propulsion system or a combination of a primary system and an auxiliary independent propulsion system. For the primary system, turboshaft cycles are always used. For the auxiliary independent system, either turboshaft, turbofan, or turbojet cycles may be used. The program includes the applicable cycles (shown in Table 2-3) from the standard library of eighty-one different generalized engine cycles developed for the VASCOMP II program. The user of the program may either select the desired engine cycle(s) from the standard library or input the characteristics of any arbitrary engine cycle he may choose.

The library engines are unrestricted in performance over their operating system range (dictated by power setting limits). However, the user, at his discretion, may include limits on engine operation by setting maximum values of fuel flow, torque, or gas generator or power turbine shaft rpm. In addition, nonlinear scaling effects of real engines may be included by input of Reynolds number-based correction factors. Degradation in performance of turboshaft engines operating at nonoptimum power turbine speed will be calculated by the program at the option of the user. The library engine cycles may thus be used with no additional input; or, by appropriate additional input, may be made to include the effects of multiple operating restrictions and other factors characteristic of real engine cycles.

During a sizing calculation, the engine cycles may be "scaled" or fixed in size. That is, if the user desires, the program will calculate the engine size required to meet the mission requirements; or, alternatively, he may input engines of specified size. In the case of helicopters employing multiple propulsion systems, the primary system may be sized to provide power to the main rotor(s) for producing lift and part of the total propulsive thrust required; and the auxiliary independent system will be sized to provide the remaining propulsive thrust or power.

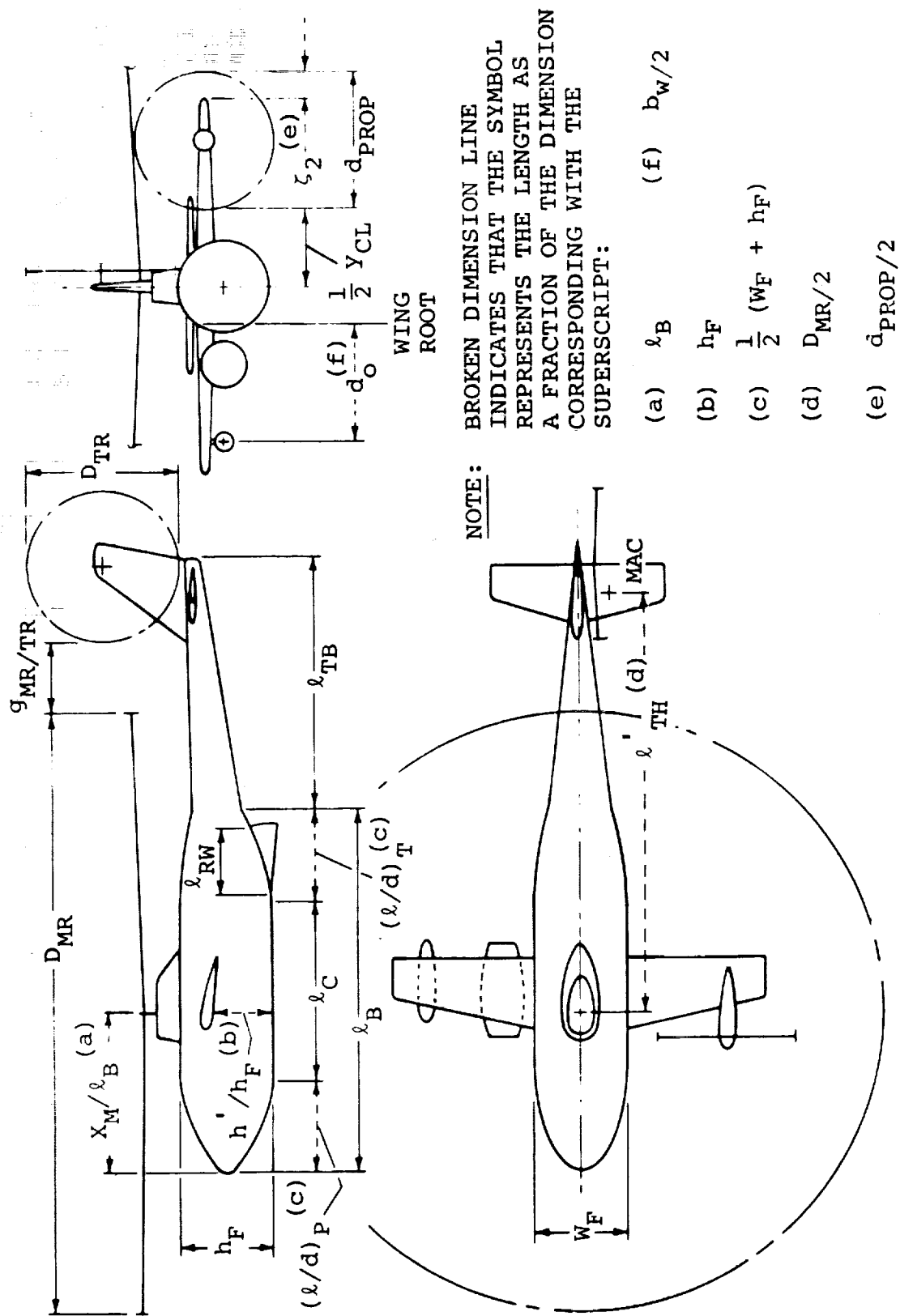
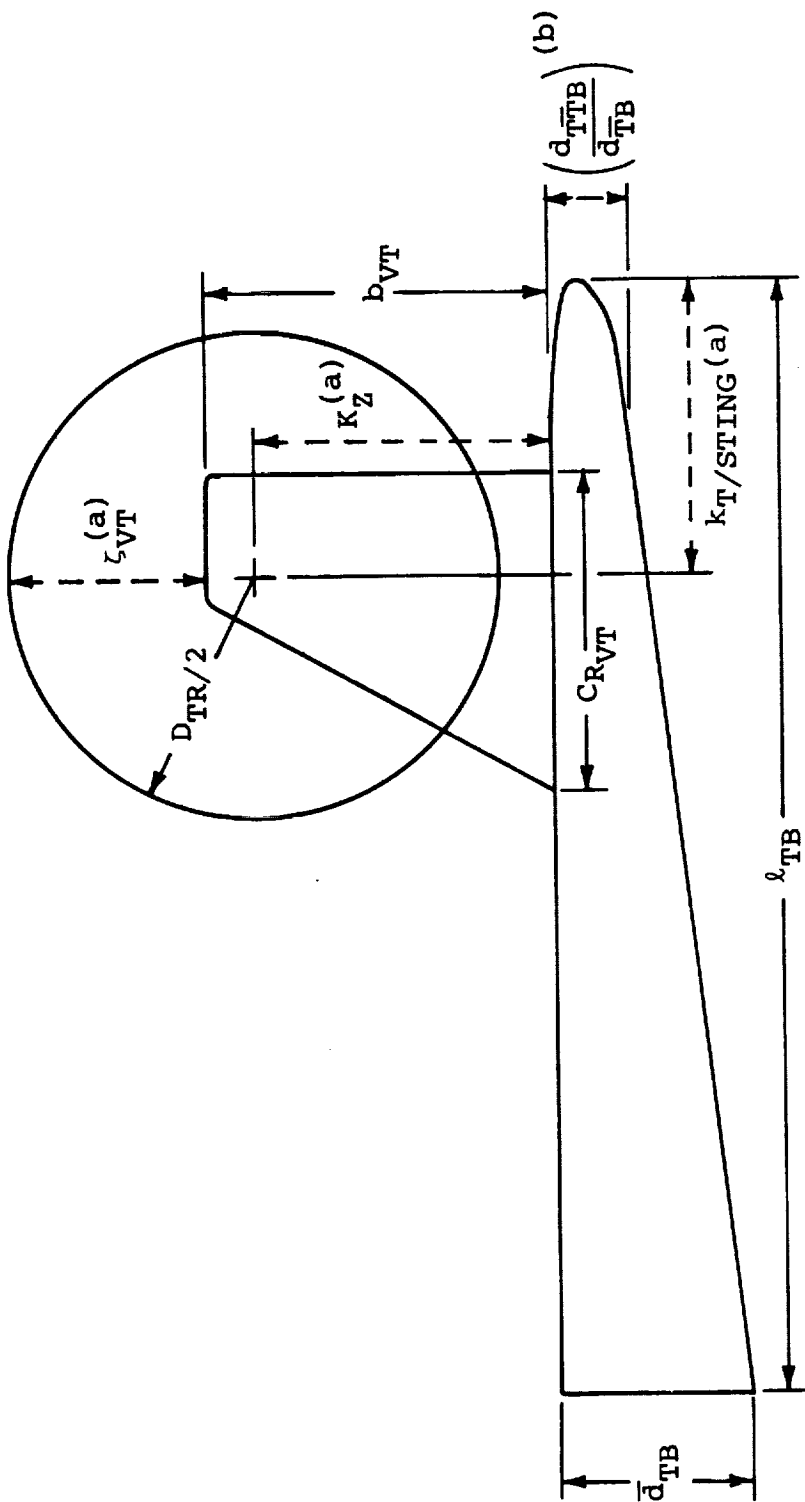


Figure 2-1. Typical Helicopter Geometry ~ Single Rotor Helicopter
(Part 1 of 2).



NOTE: BROKEN DIMENSION

LINE INDICATES

THAT THE SYMBOL

REPRESENTS THE

LENGTH AS A FRACTION

OF THE DIMENSION

CORRESPONDING WITH

THE SUPERScript:

(a) $\frac{D_{TR}}{2}$ (b) d_{TB}

$$\left. \begin{aligned} \lambda_{VT} &= \frac{C_{TV_{VT}}}{C_{RV_{VT}}} \\ AR_{VT} &= \frac{b_{VT}^2}{S_{VT}} \end{aligned} \right\}$$

Figure 2-1. Typical Helicopter Geometry ~ Single Rotor Helicopter
(Tail Boom/Tail Fin/Tail Rotor Geometry) (Part 2 of 2).

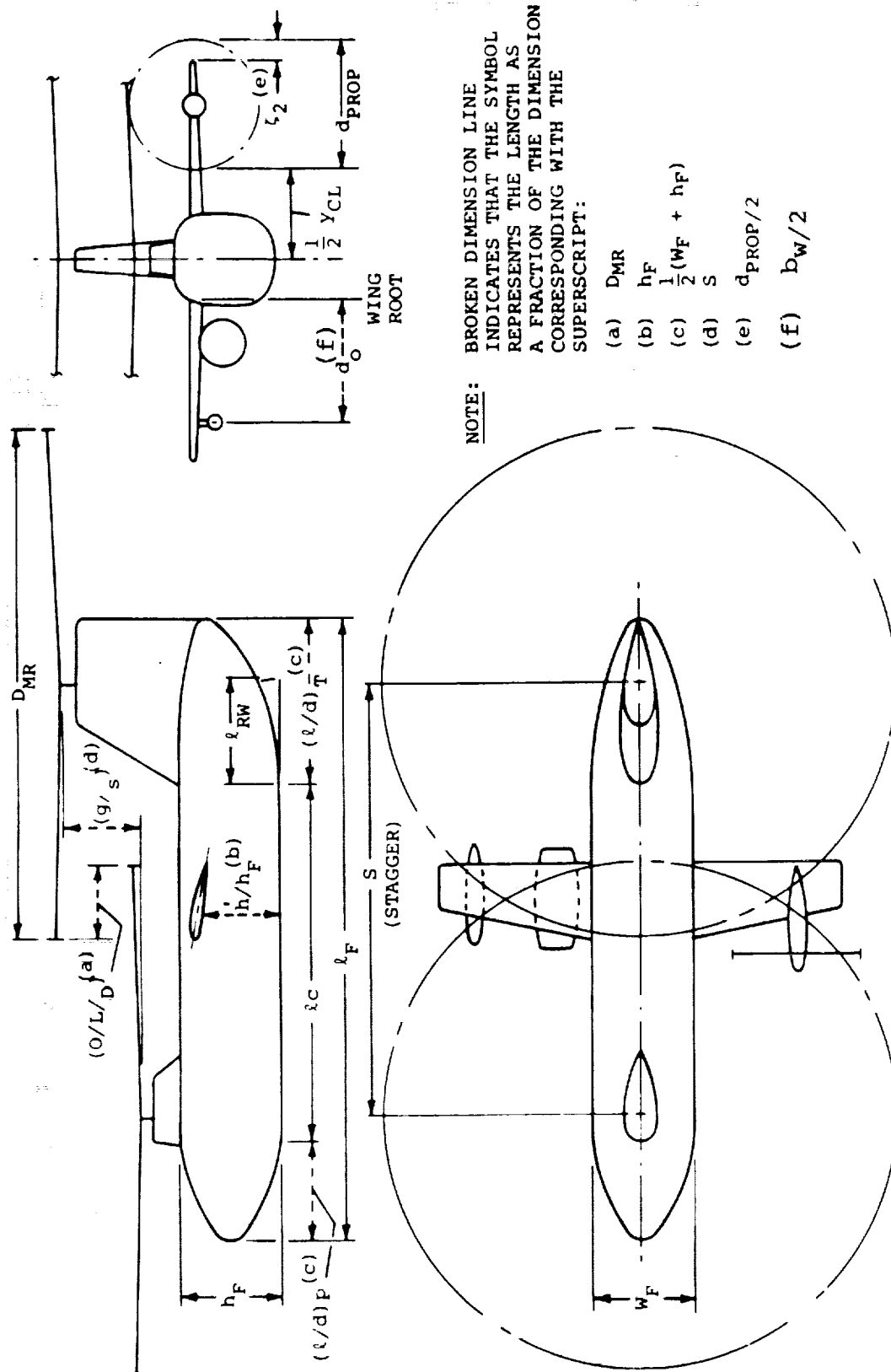
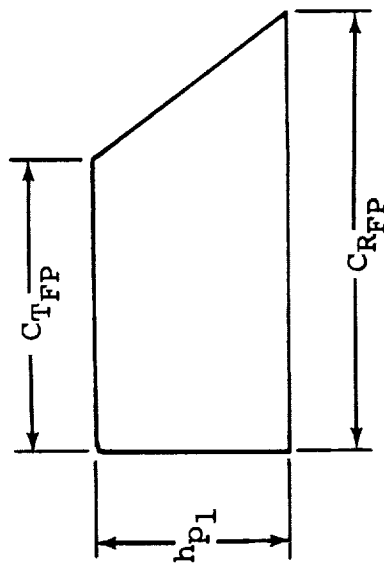


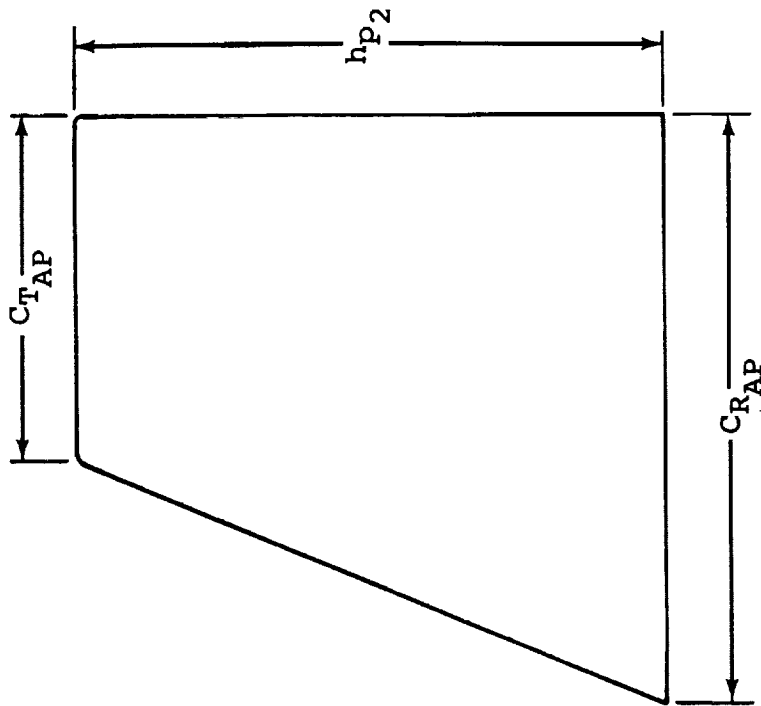
Figure 2-2. Typical Helicopter Geometry ~ Tandem Rotor Helicopter (Part 1 of 2).



FORWARD OR MAIN ROTOR PYLON

$$\lambda_{FP} = C_{T_{FP}}/C_{R_{FP}}$$

$$AR_{FP} = \frac{2h_{P1}}{C_{R_{FP}}(1 + \lambda_{FP})}$$



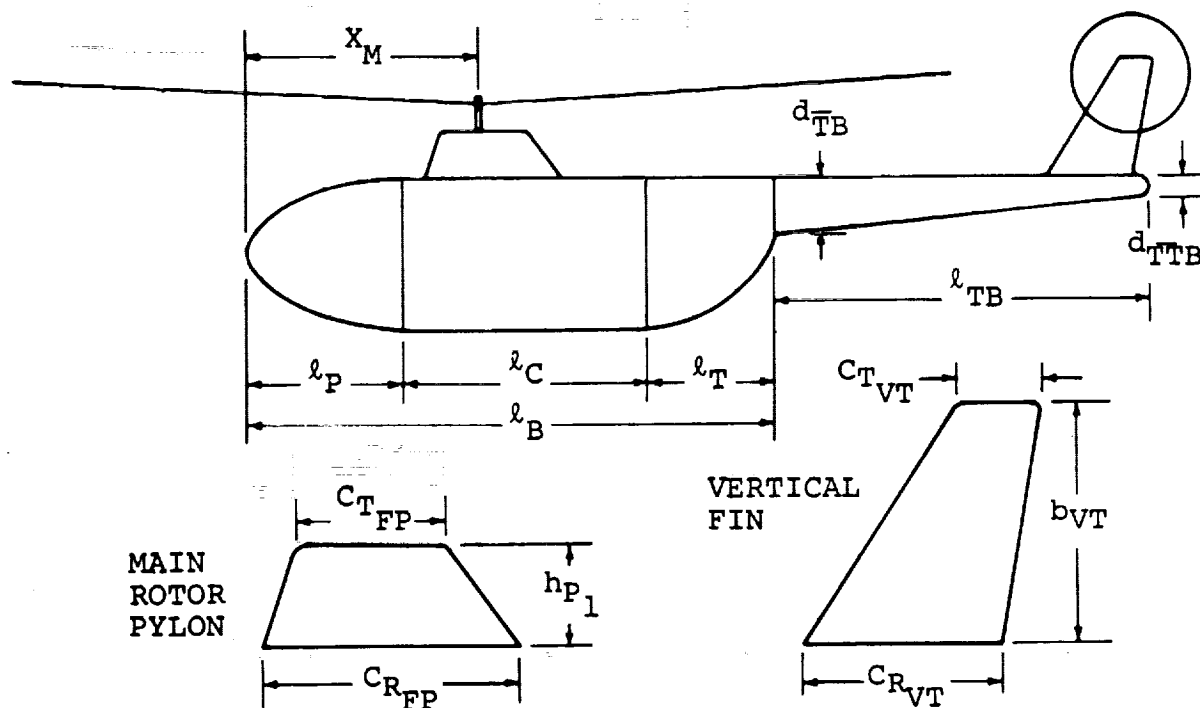
AFT ROTOR PYLON

$$\lambda_{AP} = C_{T_{AP}}/C_{R_{AP}}$$

$$AR_{AP} = \frac{2h_{P2}}{C_{R_{AP}}(1 + \lambda_{AP})}$$

Figure 2-2. Typical Helicopter Geometry ~ Tandem Rotor Helicopter
(Rotor Pylon Geometry) (Part 2 of 2).

TABLE 2-1. TYPICAL GEOMETRIC CHARACTERISTICS - SINGLE ROTOR HELICOPTERS



$$\lambda_{FP} = \frac{C_{T_{FP}}}{C_{R_{FP}}}$$

$$AR_{FP} = \frac{h_{P1}^2}{S_{FP}}$$

$$S_{FP} = \text{Planform area} \quad d_F = \frac{W_F + h_F}{2}$$

$$\lambda_{VT} = \frac{C_{T_{VT}}}{C_{R_{VT}}}$$

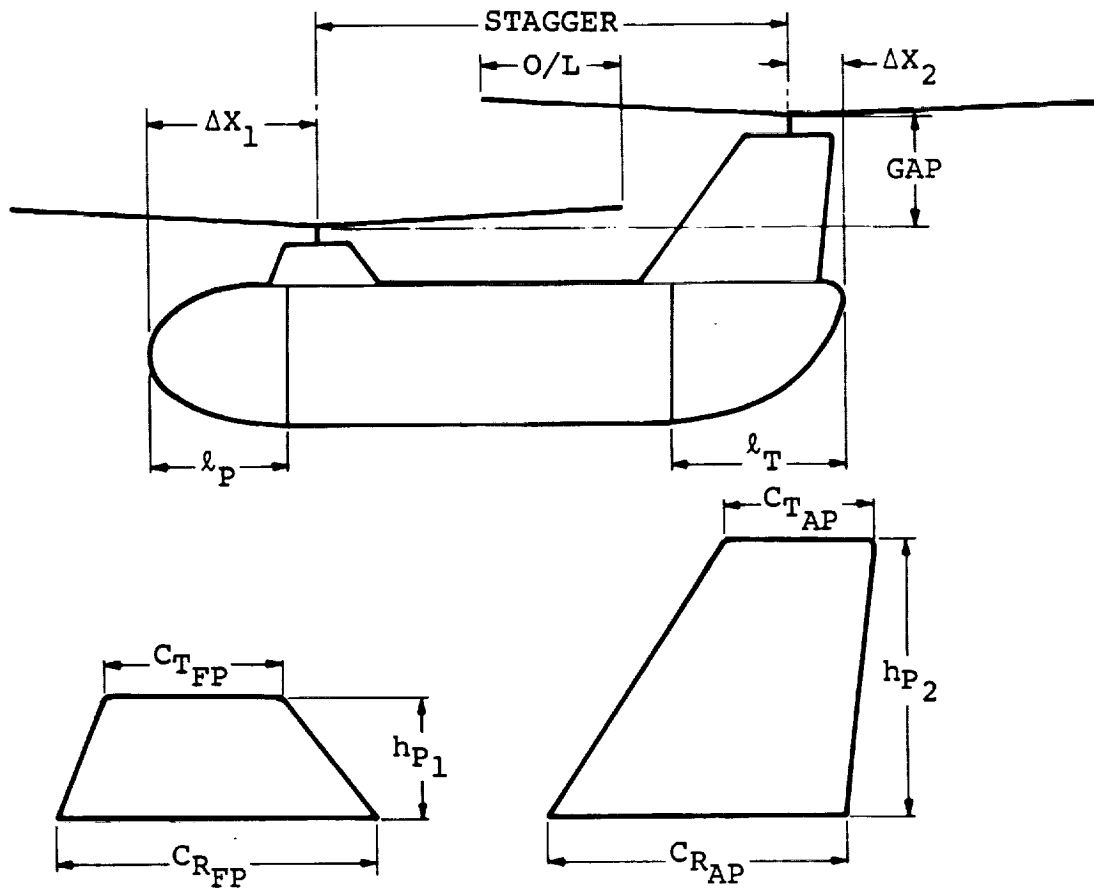
$$AR_{VT} = \frac{b_{VT}^2}{S_{VT}}$$

$$S_{VT} = \text{Planform area}$$

DATA

TYPE	AIRCRAFT	$p/\bar{d}_F$	$T/\bar{d}_F$	X_M/B	l_{TB}/d_{TB}	d_{TTB}/d_{TB}	FP	AR _{FP}	VT	AR _{VT}
"PURE" LIGHT HELICOPTERS	BOE BO-105	1.02	.75	.57	7.00	.69	.41	.30	.33	1.37
	BELL OH-58A	1.12	.71	.55	8.50	.35	.53	.15	.64	4.37
"PURE" LT. TRANSPORT HEL.	BELL OH-1B	.84	.77	.71	5.55	.32	.88	.12	.23	1.98
"PURE" MEDIUM TRANSPORT HELICOPTERS	SIK H-34A	.52	.67	.44	2.42	.42	.44	.22	.64	.96
	SIK CH-3C	.78	1.13	.44	4.45	.64	.83	.25	.75	1.72
	SIK CH-53A	.76	1.47	.42	4.38	.94	.74	.21	.62	2.14
	SIK H-37A	.60	-	.56	2.30	.47	.45	.23	.63	3.36
WINGED HELICOPTERS	BELL AH-1G	2.47	.44	.65	4.91	.42	.70	.27	.48	1.53
	SIK S-67	2.22	1.98	.62	4.17	.47	.83	.27	.48	1.90
COMPOUND HELICOPTERS	LOCK XH-51A	1.29	.98	.48	5.92	.26	.89	.33	.60	.73
	LOCK AH-56A	1.40	1.36	.62	4.43	.52	.77	.12	.46	1.29

TABLE 2-2. TYPICAL GEOMETRIC CHARACTERISTICS
TANDEM ROTOR HELICOPTERS



$$\lambda_{FP} = \frac{C_{TFP}}{C_{RFP}}$$

$$\lambda_{AP} = \frac{C_{TAP}}{C_{RAP}}$$

$$AR_{FP} = \frac{h_{P1}^2}{S_{FP}}$$

$$AR_{AP} = \frac{h_{P2}^2}{S_{AP}}$$

S_{FP} = planform area

$$d_F = \frac{W_F + h_F}{2}$$

S_{AP} = planform area

TYPE	AIRCRAFT	l_P/d_F	l_T/d_F	$O/L/D$	GAP/STAG	$\Delta X_1/l_P$	$\Delta X_2/l_T$	λ_{FP}	AR_{FP}	λ_{AP}	AR_{AP}
"PURE" LIGHT TRANSPORT HELICOPTER	B-V HUP-1	.73	1.03	.37	.08	1.00	1.00	.38	.35	.57	.78
	B-V H-21	.64	2.19	.04	.00	.89	.02	.35	.45	.23	.43
"PURE" MEDIUM - HEAVY TRANSPORT HELICOPTERS	B-V CH-46A	.71	1.53	.33	.07	1.16	.51	.63	.34	.69	.61
	B-V CH-47C	.69	1.29	.34	.12	.85	.59	1.00	1.10	.81	.53
	B-V YH-16A	.84	1.81	.37	.16	.48	.84	.34	.25	.58	.84
	B-V MOD 347	.85	1.37	.23	.15	.73	.55	.98	.43	.84	.66

TABLE 2-3. LIST OF ENGINE CYCLES

	1970	Intermediate	1980
<u>Primary Propulsion</u>			
Turboshaft			
Engine press. ratio	13, 16	13, 16, 19	13, 16, 19, 22
Turb. inlet temp.	2600°R	2900°R	3200°R
<u>Auxiliary Independent Propulsion</u>			
Turboshaft			
Engine press. ratio	13, 16	13, 16, 19	13, 16, 19, 22
Turb. inlet temp.	2600°R	2900°F	3200°R
Turbojet			
Engine press. ratio	13, 16	13, 16, 19	13, 16, 19, 22
Turb. inlet temp.	2600°F	2900°	3200°R
Turbofan			
Engine press. ratio	16, 20	16, 20, 24	16, 20, 24, 28
Turb. inlet temp.	2600°R	2900°R	3200°R
Fan bypass ratio	2, 4, 6	2, 4, 6	2, 4, 6

2.3 HELICOPTER WEIGHT SUMMARY

A detailed helicopter weight summary is provided by the program through use of statistical weight trend equations. A description of, and justification for, these equations is given in Section 4.11. Three major categories of weights are calculated: the propulsion group, the structures group, and the flight controls group.

2.4 AERODYNAMIC CHARACTERISTICS

The aerodynamic data which are calculated by the program are the helicopter drag and (in the case of winged or compound helicopters) the lift curve slope of the wing (used for calculations of the gust load factor). Drag data may be input to the program in a variety of forms including a single point value of flat plate area, drag trends, or by a detailed drag summary. Scaling effects on drag based upon Reynolds number corrections are included. Wing spanwise loading efficiency (Oswald's factor) may be either input to the program or may be program calculated.

2.5 ROTOR CHARACTERISTICS

Rotor performance may be calculated either by the short form aerodynamic performance method or by using input rotor maps. The short form method employs input rotor "cycles". Corrections for the specific rotor and helicopter configuration geometry, (e.g., blade twist, number, cut-out, rotor overlap, etc.) being analyzed are made by the program. Included with the program is a brief library of currently available "cycles" (Table 2-4 lists their pertinent characteristics).

Rotor maps may be used in two ways. In the first case, isolated rotor data derived for a specific rotor configuration is input and, as in the short form method, blade and configuration geometry corrections are applied by the program. In the second approach, a rotor map generated from total configuration rotor power data (e.g. in the case of a single rotor helicopter, this would be the sum of main and tail rotor power) is input. No corrections are applied. Thus, the particular blade and helicopter configuration geometry inherent in the data from which the map is generated is reflected unchanged in the calculated rotor performance.

TABLE 2-4. LIST OF ROTOR CYCLES

Rotor Cycle No.	Rotor Blade Planform Char.	Rotor Blade Airfoil Section Description	Rotor Blade Spanwise Airfoil Distr. Represented in Rotor Cycle	Helicopters Employing This Rotor Blade Type
1	Constant Chord	"Conventional" symmetrical NACA airfoil section	NACA 0012 from C.L. ROT to blade tip	CH-47A UH-1B SH-3A
2	Constant Chord	Cambered blade section developed by modification of NACA 230 (5 digit) series airfoil section	BV 23010-1.58 from C.L. ROT to blade tip	CH-47C MOD. 347 BO-105C
3	Constant Chord	High speed (transonic) airfoil section (s) developed from NACA 6 - series airfoil sections and optimized for maximum lift and low pitching moments.	BV VR-7 from C.L. ROT to $r/R = 0.85$ BV VR-8 at blade tip ($r/R = 1.00$)	HLH (XCH-62)

3.0 PROGRAM OPERATION

3.1 GENERAL

3.1.1 The Option Indicator

As previously described, the program has two major options and a third which is a combination of these two. The specific option to be used is selected by means of an input "option indicator" abbreviated OPTIND.

OPTIND = 1

This is an iterative routine which determines the aircraft weight, dimensions, and required power to satisfy a prescribed mission flight profile. In addition to the flight profile, certain characteristics describing the type of aircraft are specified, such as the wing aspect ratio, thickness ratio, the wing loading or disc loading, the engine cycle, etc.

OPTIND = 2 or 3

These options are used to calculate the flight performance of an aircraft for which the size is fixed. In addition to the aircraft characteristics described above, the power available, aircraft dimensions, etc. are input to the program. A flight profile is also specified. The program then calculates the performance history of the aircraft for the specified mission.

If OPTIND = 2 is selected, the aircraft gross weight is input and the fuel required to fly the specified mission is determined. This option is useful for solving many different performance problems where it is desired to constrain gross weight, such as calculating climb performance, cruise performance, or payload-range capability.

If OPTIND = 3 is selected, the operating-weight-empty is input and takeoff gross weight and required fuel load is determined. This option is useful for calculating various overload off-design weights and for determining ferry performance.

Combined Option

This option permits the user to size an aircraft for a "design-point" mission and then to calculate the off-design-point performance of the sized aircraft for a variety of additional missions. Basically, this option causes the program to run option number one (OPTIND = 1), save the sizing data generated in that option, and then input this information into the performance option (OPTIND = 2).

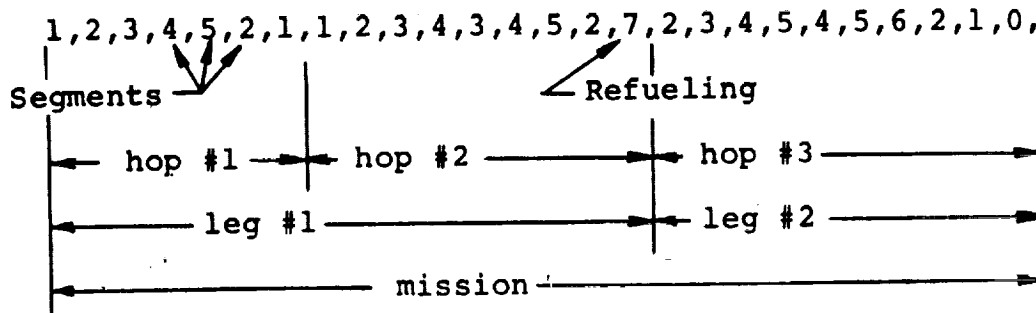
3.1.2 Description of Mission Profile

The performance calculation subprogram in HESCOMP, consisting of nine individual subroutines, permits the simulation of aircraft performance for virtually any mission flight profile. A typical performance analysis is made up of a series of elements which, in building block fashion, allows the user of the program to perform a wide variety of studies. The elements of a typical performance analysis are:

1. Segment - A segment of a mission profile is a unique portion of the mission such as a cruise or a climb. A segment starts with a set of initial conditions of one or more of the variables of state (altitude, range, weight, etc.) and ends when a terminal condition (or conditions) has been satisfied.
2. Hop - A hop is defined as a set of segments ending at some logical terminal locations (such as ground level at the desired range). Thus, a hop might consist of flying from location "A" to location "B" by means of combining the following segments: taxi, takeoff, climb, cruise, descent, landing, and taxi.
3. Leg - A leg of a mission is herein defined as a set of hops ending in a re-fueling of the aircraft. Thus, a leg might consist of flying from location "A" to "B", then to "C", at which point the aircraft is refueled.
4. Mission - A mission is defined in this program as a set of legs (or hops or segments) which satisfy some specific operational requirement. In this program, the mission is the basic element for which the aircraft is sized.
5. Case - A case is a consecutive series of missions for the same aircraft. This program permits the user to analyze a case which consists of a mission for which an aircraft is sized, followed by a different mission which the now-sized aircraft performs, followed by yet additional missions.

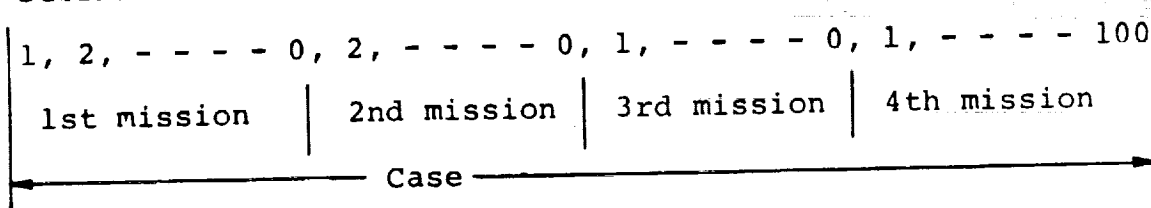
The performance calculations subprogram consists of nine individual performance segments, specified by means of an input indicator, SGTIND. The segments are taxi (SGTIND = 1), hover (SGTIND = 2), climb (SGTIND = 3), cruise (SGTIND = 4), descent (SGTIND = 5), loiter (SGTIND = 6), an increment in weight of fuel (SGTIND = 7) or payload (SGTIND = 8), and a transfer of altitude (SGTIND = 9). The end of the mission is specified by an input SGTIND = 0. An array of segment indicators is input to the program to specify the mission being studied. Thus, a typical array might be:

SGTIND =



At the end of any leg, the sum of segment fuel required to perform that leg is stored in the computer. At the end of the mission, the largest of these stored values is used to determine the aircraft sizing requirements when OPTIND = 1. An end of a case is specified by an input SGTIND = 100. Since an end-of-case is also always an end-of-mission, it is not necessary to end a case by a SGTIND = 0 followed by SGTIND = 100. SGTIND = 100 always takes precedence over SGTIND = 0. The distinction between a mission and a case is most useful when it is desired to size an aircraft for a specified mission followed by analysis of the off-design-point performance of the "sized" aircraft on other missions. As an example, with SGTIND = 1 (sizing option) the following array of SGTIND might be used:

SGTIND =



The program will size the aircraft for the first mission and then analyze the performance of the "sized" aircraft for the second, third and fourth missions. Up to 50 consecutive segments may be included in a single case, arranged in any arbitrary series of hops, legs, and missions. Up to 10 of any specific segments may be included in any case. Thus, a case might consist of several missions, each mission having several different cruise segments.

Each segment is a discrete element of the mission, independent of any other segment with the exception of the influence on the altitude, range, weight, and time. That is, the first

cruise of a case might be at cruise power at standard atmospheric conditions and the second cruise could be at best specific range for a nonstandard day.

At the start of a case, the user inputs values for initial conditions of altitude, range, weight, and time. The first segment of the case uses these values as initial boundary conditions and the segment ends at a specified terminal condition. The final values of altitude, range, weight, and time then become, in turn, the initial values for the following segment.

The final, or terminal, condition varies depending upon the segment. Terminal conditions for each segment, input by the user, are:

Taxi - increment in time

Takeoff, Hover, and Landing - increment in time

Climb - altitude at end of climb

Descent - altitude at bottom of descent and, for certain options, range at end of descent

Loiter - increment in time

Change of Fuel Weight - increment in weight and increment in time

Change of Payload Weight - increment in weight and increment in time

Transfer Altitude - final altitude

Segments 2 through 6 (takeoff, hover, and landing through loiter) require, in addition to terminal conditions on one of the variables of state, an input value for the step size to be used in the calculations. The step size specifies both the increment in the primary variable which is used in the calculations and the increment between successive printouts. Printouts occur at even integral multiples of the primary variable. Thus, if an aircraft is required to climb from a starting value of altitude of 6300 feet to a final value of 29,500 feet, and the step size is specified as 1000 feet, the program will calculate and print at 6300 feet, 7000 feet, 8000 feet, etc. to 29,500 feet. As the step size is decreased, the program accuracy improves, but the computing time lengthens.

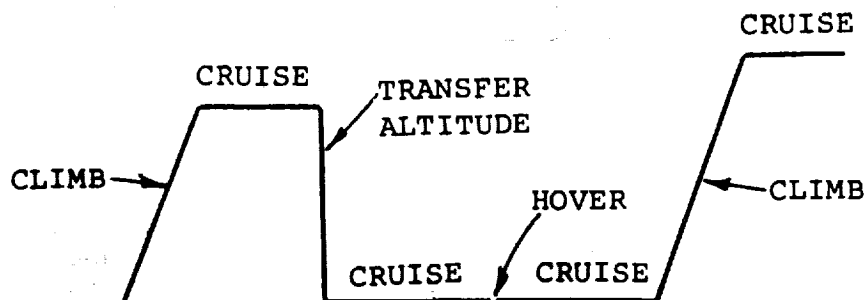
Atmospheric conditions may vary from segment to segment. For example, the first segment, a climb, may be for a standard atmosphere; the second segment, a cruise, may use a constant increment in temperature above standard; and the third segment,

another climb, may use a nonstandard temperature versus altitude table. The third atmosphere option requires a tabular input of temperature ratio versus altitude. Only one nonstandard tabular atmosphere may be used in a single case. Segments 1 through 6 (taxi through loiter) may be used to simulate an additional requirement for reserve fuel. The reserve fuel calculated in this manner is used as part of the total fuel required to size the aircraft. However, the aircraft weight is not reduced by the amount of the reserve fuel. This option is specified by inserting a value of $10 \times \text{SGTIND}$ for the particular mission segment indicator where reserve fuel is to be calculated. For example, if it is desired to calculate reserve fuel at a specified cruise condition, $\text{SGTIND} = 40$; i.e., $(\text{SGTIND} = 4) \times 10$ is input.

3.1.3 Special Flight Path Control Option

hOPTIND - This indicator will permit the user to fly a mission at the optimum altitude for best fuel consumption. The program will automatically determine the best altitude for any cruise segment which is preceded by either a climb segment or a transfer of altitude. If the cruise is preceded by a climb, the program will determine the flight altitude which minimizes the sum of the fuel for climb and cruise. If the cruise is preceded by a transfer altitude, the program will determine the altitude for the best fuel consumption during cruise only.

In addition to specifying that optimum altitude flight is desired during the mission, the user may specify a maximum altitude permitted for each cruise segment. This is specified by means of the h_{MAX} input for the preceding climb or the h_{FINAL} input for the preceding transfer altitude. The maximum altitude specification is useful in studying missions for which some of the cruise segments are to be optimized while other cruise segments are to be flown at known altitude such as the high-low-low-high mission shown below in which the low altitude segments represent sea level dashes. For this mission, the user specified $h_{\text{FINAL}} = 0$ for the transfer altitude segment.



3.1.4 Propeller Efficiency

Propeller efficiency can be calculated in three different ways for compound and auxiliary propulsion helicopters. The option chosen is specified by means of a propulsive efficiency indicator, η_{pIND} . The options range from (a) input of a set of point values of efficiency to (b) input of a prop map table to (c) automatic calculation of propeller performance. The option chosen will depend on the type of problem being studied as each of the means of calculating prop performance has features which may be desirable under certain conditions. These options are described in more detail in Section 4.7.

3.1.5 Rotor Power Required Calculation

The method most likely to be used, and certainly the most convenient, from the point of view of inputs, is the short form aerodynamic performance method. Rotor blade performance data is input in the form of "cycles", with corrections for the specific rotor and helicopter configuration under study being applied by the program.

Two types of rotor maps may be input. The first utilizes isolated rotor data derived for a specified rotor configuration, but corrected by the program for the specific rotor and helicopter configurations under study. The second uses total configuration rotor data (i.e., in the case of a single rotor helicopter, this would include both main and tail rotor power) and applies no corrections to the data.

It should be noted that the first two methods are suitable for use both in sizing and performance only calculations, since corrections for variations in rotor and helicopter configurations are applied. The third method, however, must be restricted to use only in non-sizing applications. Possible areas of use could be, for example, the case where (a) it is inconvenient for the magnitude of the particular application to generate the data required for creating a rotor cycle or generalized rotor map, or (b) in calculating the mission performance of existing helicopters (e.g. the HH-43B, WG-13, UH-2, etc.) utilizing rotor maps derived from Flight Handbooks, etc.

3.2 PROGRAM OPTIONS

Flexibility of operation and generality of approach have been accomplished by use of many optional computation paths. The path to be used is selected by the user through use of a series of input indicators. Besides the option indicator, previously described, the program indicators fall into seven categories: propulsion indicators, aerodynamics indicators, size trends indicators, mission performance indicators, flight path control

indicators, an atmosphere indicator, and an optional print indicator. The indicators and their use are described below. A summary list of all indicators and their values is included in Section 5.3.2.

3.2.1 Propulsion Indicators

AIPIND - Indicator which differentiates between compounds with and without auxiliary independent engines. AIPIND = 1 denotes a compound helicopter having a single set of engines connected both to the main rotor and auxiliary propulsion systems. AIPIND = 2 indicates a compound helicopter with independent engines for auxiliary propulsion.

ENGIND - Two different classes of cruise engines are included in the program. They are "horsepower producing" engines and "thrust producing" engines. The horsepower producing engines which are included in the standard engine library are turbo-shaft engine cycles. The thrust producing engines in the engine library are either turbojet or turbofan engines. If ENGIND = 0, a power producing cycle is selected. If ENGIND = 1, a thrust producing cycle is selected.

ESCIND - The program permits the user to size the primary engines either for takeoff conditions only or for the more critical choice of takeoff or cruise. This is specified by means of the engine sizing indicator, ESCIND. If ESCIND = 1, the program will size the engines for takeoff conditions only. If ESCIND = 2, the program will size the engines for takeoff, then cross-check the engine size required for cruise conditions, and pick the more critical of the two conditions.

FIXIND - Engines selected for aircraft being studied in the program may be either "fixed" in size or "rubberized." If the engines are "rubberized," the engine sizing subroutine calculates the maximum power or thrust of the engines required to satisfy certain specified criteria. If the engines are fixed in size, the user inputs the level of maximum power or thrust for the engines and the engine sizing subroutine is bypassed. The user specifies the option of calculation by means of the input indicator, FIXIND. If FIXIND = 0, the engines are fixed in size. If FIXIND = 1, the engine sizing subroutine is used to calculate the size of the "rubberized" engines.

FIXINDI - FIXINDI serves the same function for the auxiliary independent engines that FIXIND does for the primary engines.

POWIND - This indicator specifies the limiting power setting to be used in climb, cruise, and for engine sizing at cruise conditions: maximum (POWIND = 0), military (POWIND = 1), and normal (POWIND = 2). A separate value of this indicator is input with each climb and cruise and for engine sizing.

WDTIND, QIND, N1IND, N10IND, N2IND - These indicators specify to the program that the primary engine performance is restricted by a maximum level of fuel flow, torque, gas generator shaft rpm, gas generator referred shaft rpm, or power turbine (output) shaft rpm. An input zero value for these indicators will permit operation restricted only by power setting (turbine temperature) limits. A unity input for any of the indicators will cause the engine operation to also be restricted by a maximum level of the appropriate variable. More than one of these indicators may be set to unity at the same time, thus simulating performance of an engine operating with multiple restrictions. N2IND has a third possible value which the user may input for turboshaft engines, N2IND = 2. This input specifies that the engine is operating at a known discrete value of output shaft speed (in general, not the optimum value). If this option is used, the user inputs the level of N_{II} for each flight segment, and the program will calculate the effect on engine performance.

WDTINDI, QINDI, N1INDI, N10INDI, N2INDI - These indicators specify to the program that the auxiliary independent engine performance is restricted by a maximum level of fuel flow, torque, gas generator shaft rpm, or power turbine (output) shaft rpm. An input zero value for these indicators will permit operation restricted only by power setting (turbine temperature) limits. A unity input for any of the indicators will cause the engine operation to also be restricted by a maximum level of the appropriate variable. More than one of these indicators may be set to unity at the same time, thus simulating performance of an engine operating with multiple restrictions. N2INDI has a third possible value which the user may input for turboshaft engines, N2INDI = 2. This input specifies that the engine is operating at a known discrete value of output shaft speed (in general, not the optimum value). If this option is used, the user inputs the level of N_{II} for each flight segment, and the program will calculate the effect on engine performance.

RNOIND - The performance of real engines is sensitive to scaling effects. That is, doubling the maximum static power of the engine at sea level for standard atmospheric conditions by increasing the physical size of the engine will not cause a corresponding doubling of the power at other operating conditions. This nonlinear behavior is due to the influence of variations in the Reynolds number at the compressor inlet. RNOIND permits these effects to be accounted for on turboshaft engines through use of an input table of a correction factor on power available. If the indicator is set to unity, the tabulated correction factor may be input and will be used by the program to account for scaling effects. A zero input for the indicator will cause the program to assume that perfect scaling occurs.

RNOINDI - This indicator serves the same purpose for the auxiliary independent engines as RNOINDI serves for the primary engines.

ROTIND - Controls the selection of the rotor performance computation method. If ROTIND = 1, rotor performance is calculated by the short form aerodynamic performance method (requiring the input of a rotor "cycle"). If ROTIND = 2, a rotor map is input with corrections being applied by the program for the specific rotor and helicopter configuration geometry being studied. If ROTIND = 3, a rotor map is input, with no corrections being applied.

η_p IND - This indicator permits the user to select one of three different methods for predicting propeller performance for compound and auxiliary propulsion helicopters. If η_p IND = 0, the user can specify a set of point value efficiencies for each climb and descent and a table of efficiency versus Mach number for cruise and loiter. An input of η_p IND = 1 will permit the user to load in a propeller performance map to be used during climb, cruise, and loiter while an input of η_p IND = 2 will permit use of an automatic subroutine within the program for calculating prop performance. It is anticipated that this latter option will be used for the majority of sizing and performance studies. The input prop map option will typically be used in cases where detailed test data is available on prop performance and it is desired to closely represent a specific propeller. The first option, permitting input of point values, is most useful for sensitivity studies or where propeller choice has not yet been made and only representative values of efficiency are desired. A more detailed discussion of these options is contained in Section 4.7.

3.2.2 Aerodynamics Indicators

DRGIND - The method of determining the total parasite drag of the helicopter is specified to the program by means of the indicator DRGIND. If DRGIND = 1, configuration parasite drag is built up in component fashion, with Reynolds number scaling. If DRGIND = 2, the parasite drag is calculated from a parasite drag trend derived from the inputs (GW/Fe) and K_{FED} .

OSWIND - The span loading efficiency factor (Oswald's efficiency factor) may be calculated by the program from an approximate relationship as a function of wing aspect ratio. If the user prefers, he may input a fixed value of the efficiency factor to the program. An input of OSWIND = 0 permits the user to input a fixed value for efficiency. An input of OSWIND = 1 will cause the program to use the approximate equation to calculate the value for efficiency.

3.2.3 Size Trends Indicators

APHIND - The aft rotor pylon height of a tandem rotor helicopter is specified by use of this indicator. If APHIND = 1, aft pylon height is input directly in feet. If APHIND = 2, the tandem rotor gap/stagger (g/s) ratio is input and aft pylon height is sized accordingly.

AUXIND - Four versions of both the single and tandem rotor helicopter may be specified through this indicator. They are: a pure helicopter (AUXIND = 1), a winged helicopter only (AUXIND = 2), an auxiliary propulsion helicopter, only (AUXIND = 3), and a compound (wings and auxiliary propulsion) helicopter (AUXIND = 4).

b_wIND - For a configuration having wings, this option determines the manner in which wing span is calculated during the sizing process. If b_wIND = 1, wing span/rotor diameter ratio (b_w/D) is input. If b_wIND = 2, wing aspect ratio (AR) is input. If b_wIND = 3 (used when dealing with wing-mounted propellers) wing span is determined from propeller tip/fuselage clearance considerations.

CNFIND - This indicator specifies the helicopter configuration to be analyzed. These are: the single rotor helicopter (CNFIND = 1) and the tandem rotor helicopter (CNFIND = 2).

FDMIND - Determines the manner in which a tandem rotor helicopter fuselage is sized. If FDMIND = 1, tandem rotor overlap ((O/L)/D) and forward and aft rotor positions ($\Delta X_1/l_p$, $\Delta X_2/l_T$) are specified. If FDMIND = 2, overlap and cabin length (l_c) are input. If FDMIND = 3, cabin length and forward and aft rotor positions are input.

HTIND - Permits the user to input fixed-size horizontal tail surfaces to the program or, optionally, to have the program calculate the tail surface size based upon an input tail "volume" coefficient. If HTIND = 0, the program will assume no horizontal tail exists. If HTIND = 1, the tail area may be input directly. If HTIND = 2, the program will calculate the size based upon a tail "volume" coefficient.

MRPIND - Specifies the placement of the main rotor of a single rotor helicopter on its fuselage. If MRPIND = 0, the user inputs directly the main rotor position (aft of the nose) as a fraction of body length (X_M/l_B). If MRPIND = 1, the program does a simple mass balance calculation and determines the rotor position relative to the aircraft cg. If MRPIND = 2, the same procedure is carried out as with MRPIND = 1 with the exception that the program assumes the auxiliary drive system, propeller, and auxiliary independent engines (if any) to be located on the wing.

RDMIND - Specifies manner in which main rotor is sized. If RDMIND = 1, main rotor diameter and solidity are input directly. If RDMIND = 2, disc loading and solidity are input, diameter is calculated. If RDMIND = 3, diameter and C_T/σ are input, solidity is calculated. If RDMIND = 4, disc loading and C_T/σ are input and both diameter and solidity are calculated.

SWIND - Specifies options available for wing sizing. These are: wing area input directly (SWIND = 1), wing area sized based on an input wing loading (SWIND = 2), and wing area sized by rotor/wing maneuver requirements (SWIND = 3).

TRDIND - Determines manner in which tail rotor diameter is sized. If TRDIND = 1, tail rotor diameter is calculated from a trend of DMR/DTR contained in the program. If TRDIND = 2, tail rotor diameter is input directly. If TRDIND = 3, a value of net tail rotor disc loading, $(T/A)_{NET}$, is input and tail rotor diameter is determined through an iterative procedure.

TRSIND - Tail rotor solidity sizing indicator. If TRSIND = 1, tail rotor solidity is input directly. If TRSIND = 2, C_T/σ is input and tail rotor solidity is sized based on either hover-antitorque or hovering turn requirements.

VTFIND - Vertical tail area sizing indicator. If VTFIND = 1, vertical tail size is based on input values of aspect ratio (AR_{VT}) and tail fin/tail rotor overlap (h_{VT}). If VTFIND = 2, tail fin/tail rotor overlap and directional stability requirements, (sufficient tail area to counteract main rotor torque in cruise flight, if tail rotor is lost), dictate vertical tail area. If VTFIND = 3, the same requirements must be met as with VTFIND = 2, with the exceptions that AR_{VT} is specified and tail fin overlap is calculated along with vertical tail area.

XMSNIND - Indicator that controls drive system transmission sizing. If XMSNIND = 1, main, tail and auxiliary drive system ratings are specified as a fraction of primary engine installed power (in the case of a compound helicopter with auxiliary independent propulsion, the auxiliary independent drive system rating is specified as a fraction of the auxiliary independent engine installed power). If XMSNIND = 2, main, tail, and auxiliary drive system ratings are specified at a fraction of the power required to hover or cruise at design conditions (more critical of the two conditions is selected). If XMSNIND = 3, the same applies as in the case where XMSNIND = 2, except the most critical of the two design conditions is compared to the drive system rating required at an alternate payload/gross weight hover at the design point conditions, the most critical of these three conditions is selected.

3.2.4 Mission Performance Indicators

CLMIND - Four types of climb calculations are permitted: maximum rate of climb (CLMIND = 1), constant equivalent airspeed (CLMIND = 2), constant Mach number (CLMIND = 3), and constant true airspeed (CLMIND = 4).

CRSIND - Six types of cruise missions are included in the program: cruise at fixed cruise power (CRSIND = 1), cruise at constant true airspeed (CRSIND = 2), cruise at airspeed for best specific range, (CRSIND = 3), cruise at the speed for 99% of best specific range (CRSIND = 4), cruise-climb (constant W/δ) at the speed for best specific range (CRSIND = 5), or cruise-climb at the speed for 99% of best specific range (CRSIND = 6).

DESIND - Twelve different descent paths may be calculated by the program. They are of three major types: descent at constant true airspeed (TAS) (DESIND = 1), descent at constant Mach number (DESCIND = 3). Four variations of each of these major types of descent are specified by RMAXND. It should be noted that there are no idle power or autorotative descent options available. However, depending on the descent flight conditions specified, it is possible to operate on an autorotative descent boundary (see Section 4.12.5) during a descent.

RMAXND - Used in conjunction with DESIND to specify types of descent. If RMAXND = 0, the descent flight path ends at a specified terminal range (cruise segment must be input previous to descent). If RMAXND = 1, the program checks the specified terminal range, and, if the predicted flight path will end beyond the specified terminal range value, a spiral descent path is assumed at that point; if the predicted flight path ends before reaching the specified terminal range point, the program prints "SHALLOWER DESCENT REQUIRED". If RMAXND = 2, the descent ends at a specified minimum altitude, terminal range requirement not considered. If RMAXND = 3, the fuel used and time required for descent are calculated but no range credit given (i.e., spiral descent path).

SGTIND - The mission profile flown by the aircraft may be made up of an arbitrary sequencing of nine discrete profile segments. The segment selected is specified by means of the segment indicator, SGTIND. The segments are: taxi (SGTIND = 1), takeoff, hover and landing (SGTIND = 2), climb (SGTIND = 3), cruise (SGTIND = 4), descent (SGTIND = 5) loiter (SGTIND = 6), a change of fuel weight (SGTIND = 7), a change of payload weight (SGTIND = 8), and a transfer of altitude (SGTIND = 9). By appropriate sequencing of the input values for the segment indicator, the mission profile may be made up of any arbitrary combination of these nine discrete elements. The mission is terminated by an input value for segment indicator = 0. NOTE: Segments 1 through 6 can be used for reserve fuel calculations (gross weight reset following segment) by inputting 10 times SGTIND, i.e., SGTIND = 10, 20, 30, 40, 50, or 60.

TOLIND - The indicator TOLIND is input with each takeoff, hover, and landing segment and dictates the manner in which power is calculated. If TOLIND = 1, the user inputs required thrust-to-weight ratio and vertical rate of climb (VR/C). If TOLIND = 2, the user inputs required fractions of maximum power and vertical rate of climb (T/W ratio is computed). Both TOLIND = 1 and 2 options are calculated, based on the assumptions of hover-out-of-ground effect. If TOLIND = 3, the option is the same as 1, but the analysis includes hover-in-ground effect factors. If TOLIND = 4, the option is the same as 2, but the analysis includes hover-in-ground effect factors.

WGTIND - The change fuel and change payload segments may be used to simulate refueling, unloading or loading of passengers, or a fuel drop. There is no restriction on the amount of fuel or payload which may be removed at any point in the mission. However, during a sizing run, it would be undesirable to increase the aircraft weight (by adding fuel or payload) to a value which exceeds the initial gross weight of the aircraft. This is because the design gross weight, upon which the subsystem weights depend, is assumed to be the same as the initial gross weight at the start of the mission. During a performance run (OPTIND = 2), this restriction does not apply and the user is given the option of overloading the aircraft at any point of the mission. If WGTIND = 0, the program will not permit the maximum weight to exceed the design gross weight. This is useful if it is desired to refuel to capacity at some point in the mission. If WGTIND = 1 (and if the performance option is being run), the program will permit the aircraft weight to exceed the design gross weight. This is useful for parametric performance studies. For example, the user can specify an array of SGTIND = 7, 4, 0, 7, 4, 0, 7, 4, 0, up to 7, 4, 100. When this is done, the program will calculate the performance in cruise at a series of different aircraft weights. The "7" segment is used to increment the design gross weight to any value of weight desired for the following cruise.

3.2.5 Flight Path Control Indicators

hOPTIND - By inputting hOPTIND = 1.0, the program will automatically determine the cruise altitude for minimum fuel consumption for any cruise which is preceded by a climb or a transfer altitude. For cruise segments which are preceded by a climb, the program will find the cruise altitude for which the sum of climb fuel and cruise fuel is minimized. The user can also specify a maximum permissible altitude for each cruise segment. If hOPTIND = 0 is input, the program will not do an optimum altitude search for the cruise segments.

3.2.6 Atmosphere Indicator

ATMIND - The atmosphere for each individual mission profile segment and for the engine sizing calculations may be either a standard or nonstandard atmosphere. Thus, the climb may be run

on a nonstandard atmosphere followed by a cruise for standard day conditions. Three options, one for standard atmosphere, the other two for a nonstandard atmosphere are available. For the performance calculations, the type of atmosphere to be used is specified to the program by means of the atmosphere indicator, ATMIND. If ATMIND = 0, the program will use a standard atmosphere. ATMIND = 1 specifies a nonstandard, constant increment in temperature above standard while ATMIND = 2 specifies a nonstandard atmosphere requiring a tabular input of temperature ratio versus altitude.

3.2.7 Optional Print Indicator

Two different forms of printout are available for the mission performance data. By setting OPTIONAL PRINT INDICATOR = 0, a standard printout will occur. This consists of time, range, fuel used, aircraft weight, pressure altitude, true airspeed, primary engine turbine temperature, an engine code which specifies the condition which is dictating the primary engine operating point, and a power fraction which is the instantaneous fraction of maximum power which is being used. These data are printed for all performance segments. In addition, depending upon which segment is being used, the standard printout will include such parameters as rate of climb, equivalent airspeed, specific range, flight path angle, etc. More detailed data may be obtained by setting the OPTIONAL PRINT INDICATOR = 1.0. The data printed will then include main rotor power and tip speed, tail rotor power and tip speed, auxiliary propulsion power and propeller tip speed, primary and auxiliary engine fuel flows, etc. The printout available from the program is described in more detail in Section 6.1.4.

3.3 PROGRAM FLOW

Figure 3-1 indicates, conceptually, the operation of the program. Program flow is monitored by a general control loop which controls the operation of a series of peripheral programs. These include eighteen minor subroutines, four major subroutines, a major subprogram, and a library of engine cycle data, and rotor "cycle" data. The characteristics of these routines are summarized in Table 3-1.

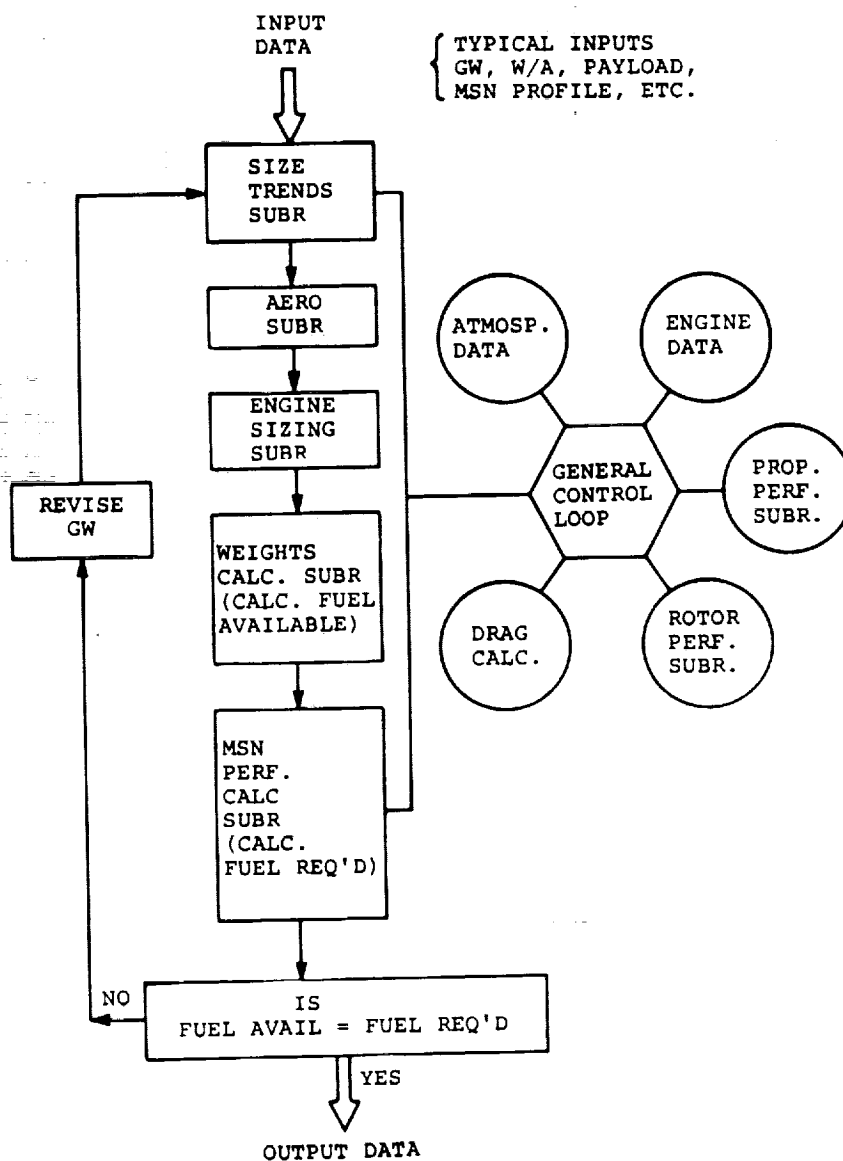


Figure 3-1. Sketch of Program Geometry.

TABLE 3-1. SUMMARY OF SUBROUTINES

ROUTINE	CALLED BY	PURPOSE
Main Control Loop (MAIN)	---	Monitors program operations, checks convergence and calculates gross weight during iterative sizing option (OPTIND = 1)
Minor Subroutines:		
Atmosphere (ATMOS)	Performance subroutines with SGTIND = 1-6, engine sizing and size trends	Calculates atmospheric density, pressure, temperature and speed of sound
DESPOW	Descent subroutine	Calculates power required for descent
Drag Calculations (DRAG)	Performance subroutines with SGTIND = 1-6, and engine sizing	Calculates aircraft propulsive force coefficient (C_x)
POWAVL	Performance subroutines with SGTIND = 1-6, engine sizing and size trends	Calculates power and fuel flow available for primary turboshaft engines by determining the most critical operating restrictions
POWAVI	Same as POWAVL	Calculates power and fuel flow available for auxiliary independent turboshaft engines by determining the most critical operation restrictions
POWREQ	Same as POWAVL	Calculates fuel flow for primary turboshaft engines when power required is less than power available.

TABLE 3-1. CONTINUED

ROUTINE	CALLED BY	PURPOSE
POWRQI	Same as POWAVI	Calculates fuel flow for auxiliary independent turboshaft when power required is less than power available
ENGL	POWREQ, POWAVL	Calculates power available for primary turboshaft engines at any specified turbine temperature, including effects of Reynold's number and operation at nonoptimum NII
ENGLI	POWRQI, POWAVI	Calculates power available for auxiliary independent turboshaft engines at any specified turbine temperature, including effects of Reynold's number and operation at nonoptimum NII
THRAVL	Same as POWAVL	Calculates thrust and fuel flow available for turbofan and turbojet engines by determining most critical operating restriction
THRREQ	Same as POWAVL	Calculates fuel flow for turbofan and turbojet engines when thrust required is less than thrust available
POWER	Performance Subroutines with SGTIND = 3,4,5,6 & engine sizing and size trends	Calculates propeller power required when thrust and advance ratio are known
POWERI	Same as Power	Calculates propeller power (for auxiliary independent propulsion system) required when thrust and advance ratio are known
PRINT 1 and PRINT 2	MAIN Subroutine	Controls printout of data generated in sizing when OPTIND = 1.0

TABLE 3-1. CONTINUED

ROUTINE	CALLED BY	PURPOSE
ROT LIM	Performance subroutines with SGTIND = 2,3,4,5&6	Checks main rotor operating conditions (C_t/σ and C_x/σ) to see if rotor limit has been exceeded.
ROT POW	Performance subroutines with SGTIND = 2,3,4,5,6 and engine sizing and size trends	Calculates rotor power required (both main and tail rotors)
SCRIBE	Performance subroutines with SGTIND = 4 & 6	Controls printout when detailed print option is desired (OPTIONAL PRINT INDICATOR = 1.0)
THRUST	Same as POWER	Calculates propeller thrust available when power and advance ratio are known
Major Subroutines:		
Aerodynamics (AERO)	Main	Calculates wing spanwise loading efficiency and a series of coefficients which are used by the drag subroutine to calculate aircraft drag
Engine Sizing (ENG SZ)	Main	Calculates engine size (power or thrust) required to meet mission requirement
Size Trends (SIZ TR)	Main	Calculates aircraft dimensions which are required for weight estimate and drag calculation
Weight Trends (WG HTR)	Main	Calculates aircraft weight summary including propulsion, structures, flight controls and fuel available

TABLE 3-1. CONTINUED

ROUTINE	CALLED BY	PURPOSE
Major Subprogram:		
Performance Calculations (PRFRM)	Main	Monitors program flow during calculation of mission performance and calculates total fuel required at end of mission
Performance Subroutines		
Taxi (TAXI)	Performance subprogram	Calculates taxi performance
Takeoff, Hover & Landing (TOHL)	Performance subprogram	Calculates takeoff, hover or landing performance
Climb (CLIMB)	Performance subprogram	Calculates climb performance
Cruise (CRUSL, 2, 3)	Performance subprogram	Calculates cruise performance
Descent (DSCNT)	Performance subprogram	Calculates descent performance
Loiter (LOITR)	Performance subprogram	Calculates loiter performance
Change Fuel Weight (CHGFW)	Performance subprogram	Adds or subtracts fuel to aircraft
Change Payload (CHGPL)	Performance subprogram	Adds or subtracts payload to aircraft
Transfer Altitude (TRALT)	Performance subprogram	Changes altitude

4.0 DETAILED PROGRAM DESCRIPTION

4.1 MAIN CONTROL LOOP

Figure 4-1 is a flow chart of the main control loop for the computer program. In the sizing option (OPTIND = 1), the program iterates on the aircraft gross weight until the fuel available and the fuel required are equivalent within a specified tolerance. If OPTIND = 2 or 3, the program bypasses the size trends, engine sizing, and weight trends subroutines. If OPTIND = 3, the program iterates to determine the takeoff weight and fuel required to fly a specified mission.

4.1.1 Input Card Setup

The first five columns of an input card contains information used by the input routine LOADER. A card with 77777 punched in the first five columns indicates a title card follows. The following card is an alpha-numeric title card with information in columns seven through seventy-eight as shown on the input sheets in the User's Manual. All input data are assigned a unique location in the input data file. This is indicated by the location number of each variable on the input sheets in the User's Manual. Up to five variables may be input on a card. Columns 1 through 4 contain the location number of the first variable on the card and column five the number of variables on the card. A card with 88888 punched in the first five columns indicates the end of data for that case and starts program execution. A card with 99999 in the first five columns indicates the end of the run and causes program termination. Cases can be stacked in the following manner.

77777	Card
	Title Card
	Data Cards
88888	Card
77777	Card
	Title Card
	New Data Cards
88888	Card
99999	Card

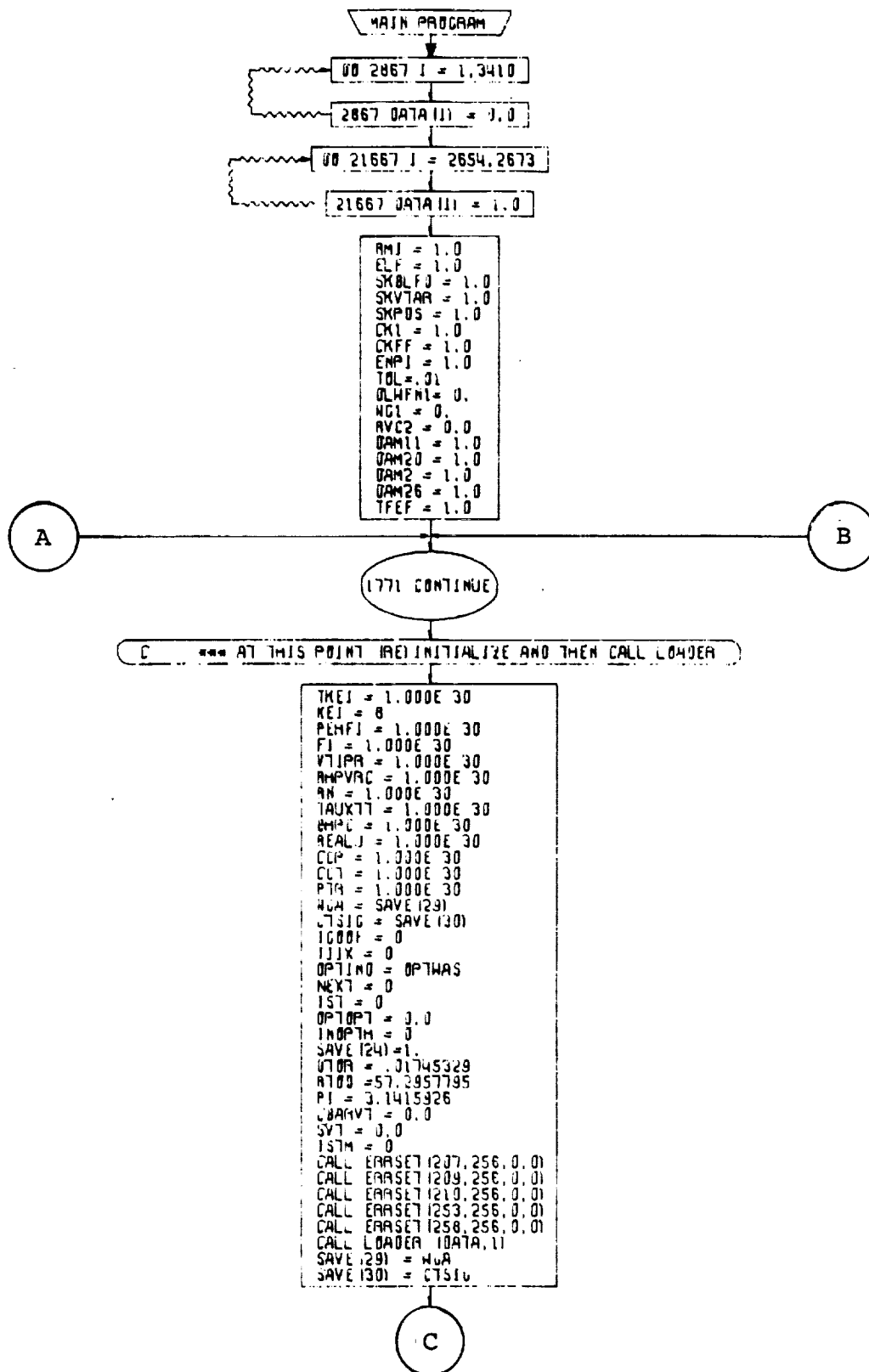


Figure 4-1. Main Control Loop (MAIN) Flow Chart (Part 1 of 13).

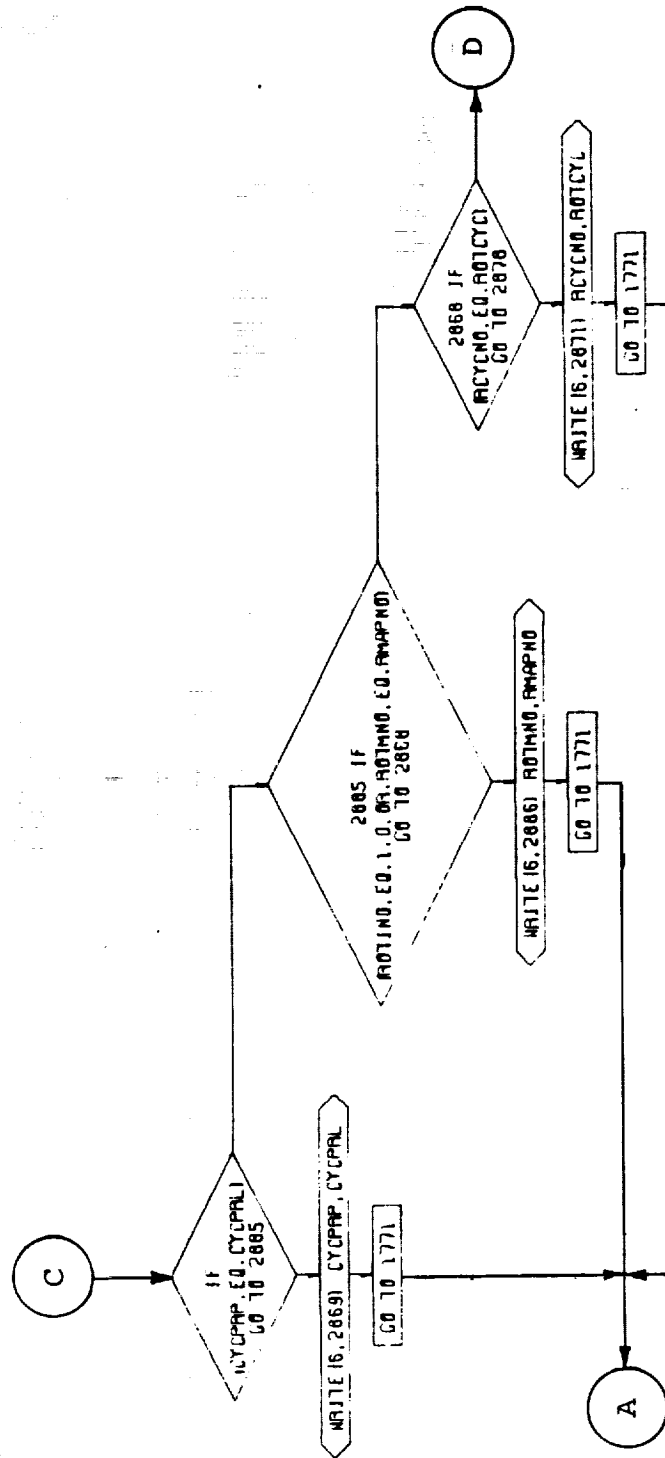


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 2 of 13).

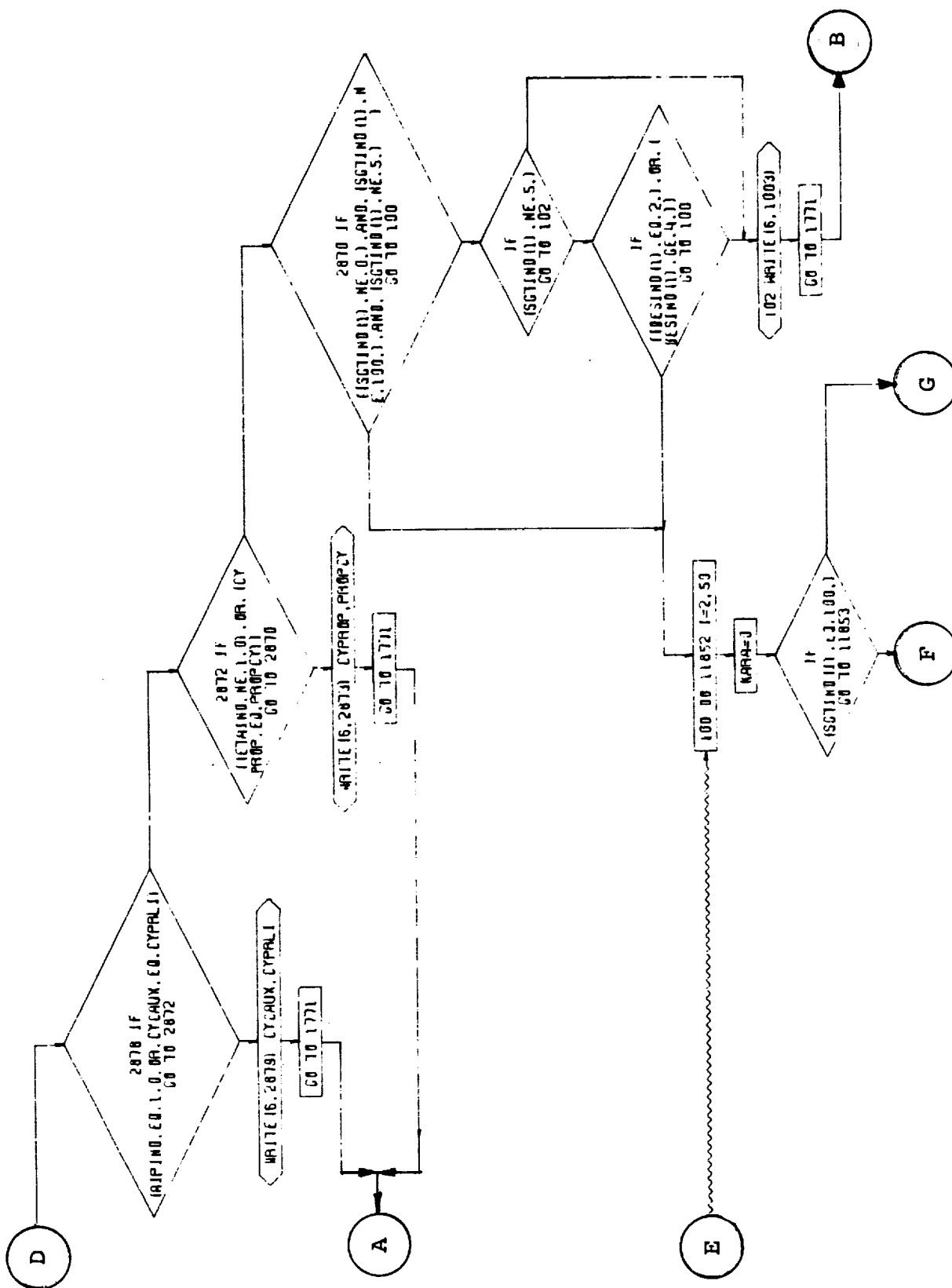


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 3 of 13).

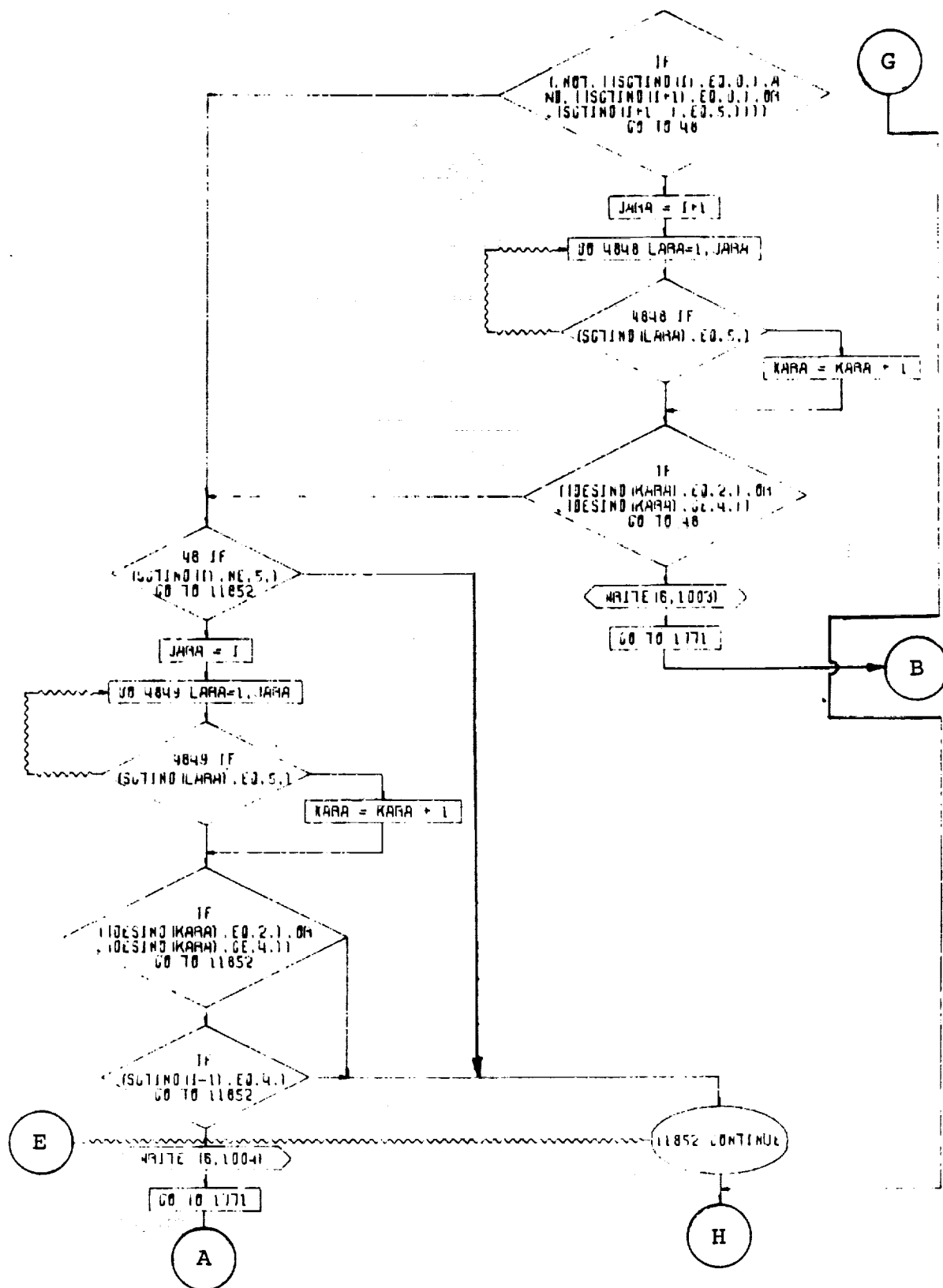


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 4 of 13).

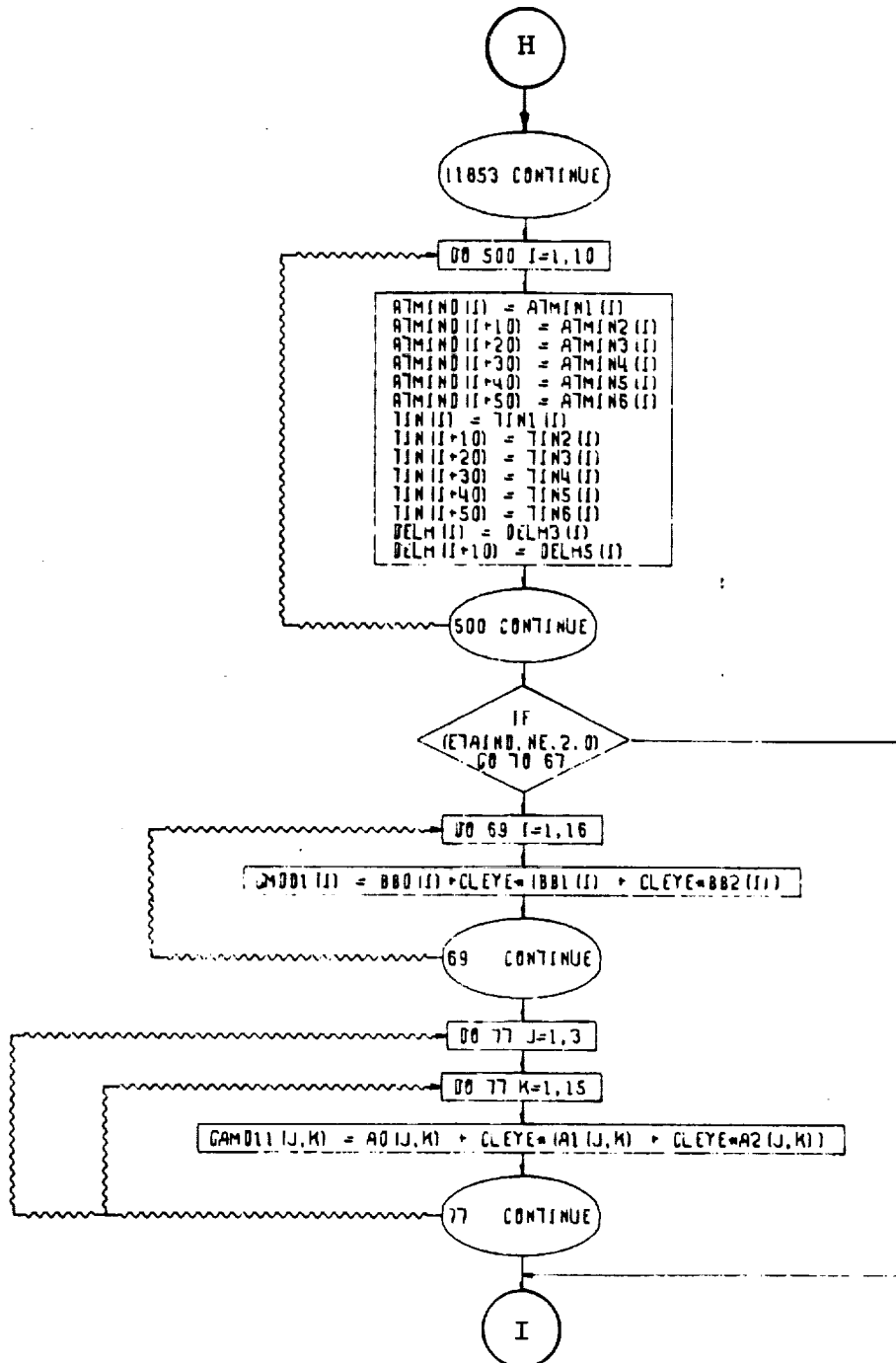


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 5 of 13).

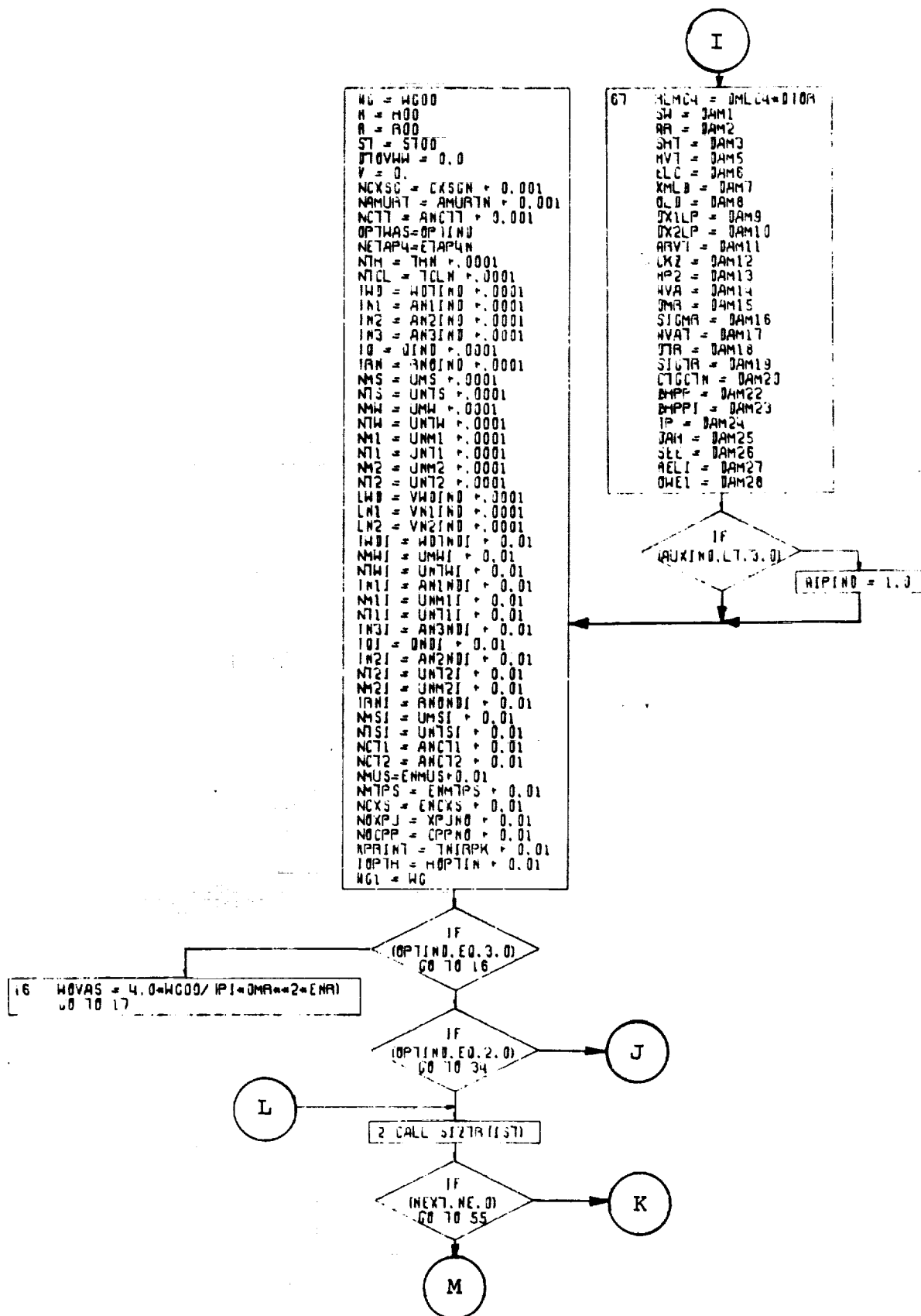


Figure 4-1. Main Control Loop (MAIN) Flow Chart (Part 6 of 13).

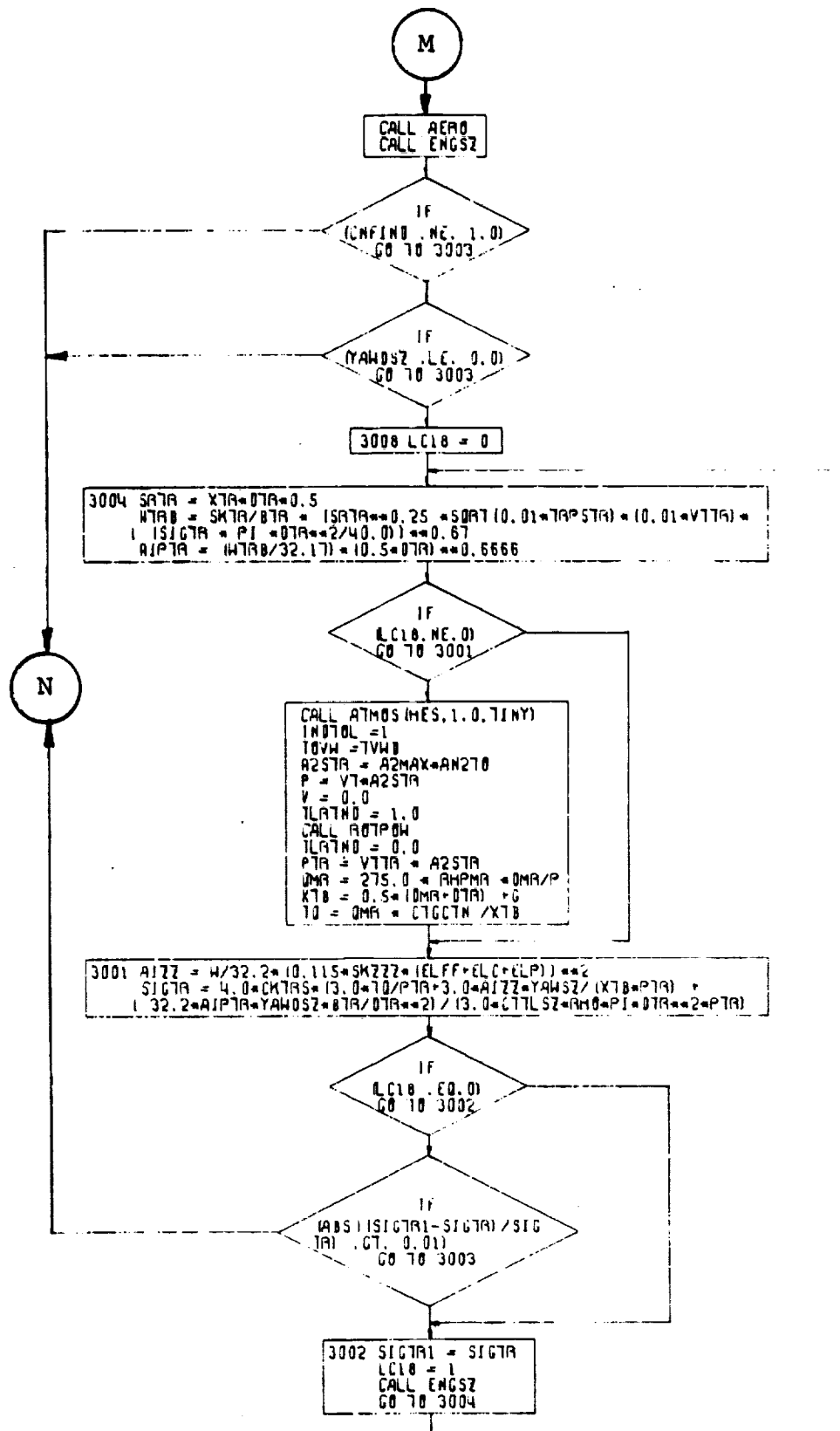
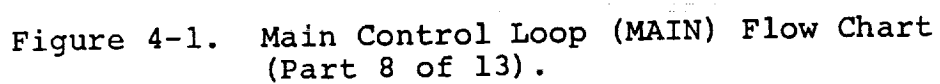


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 7 of 13).



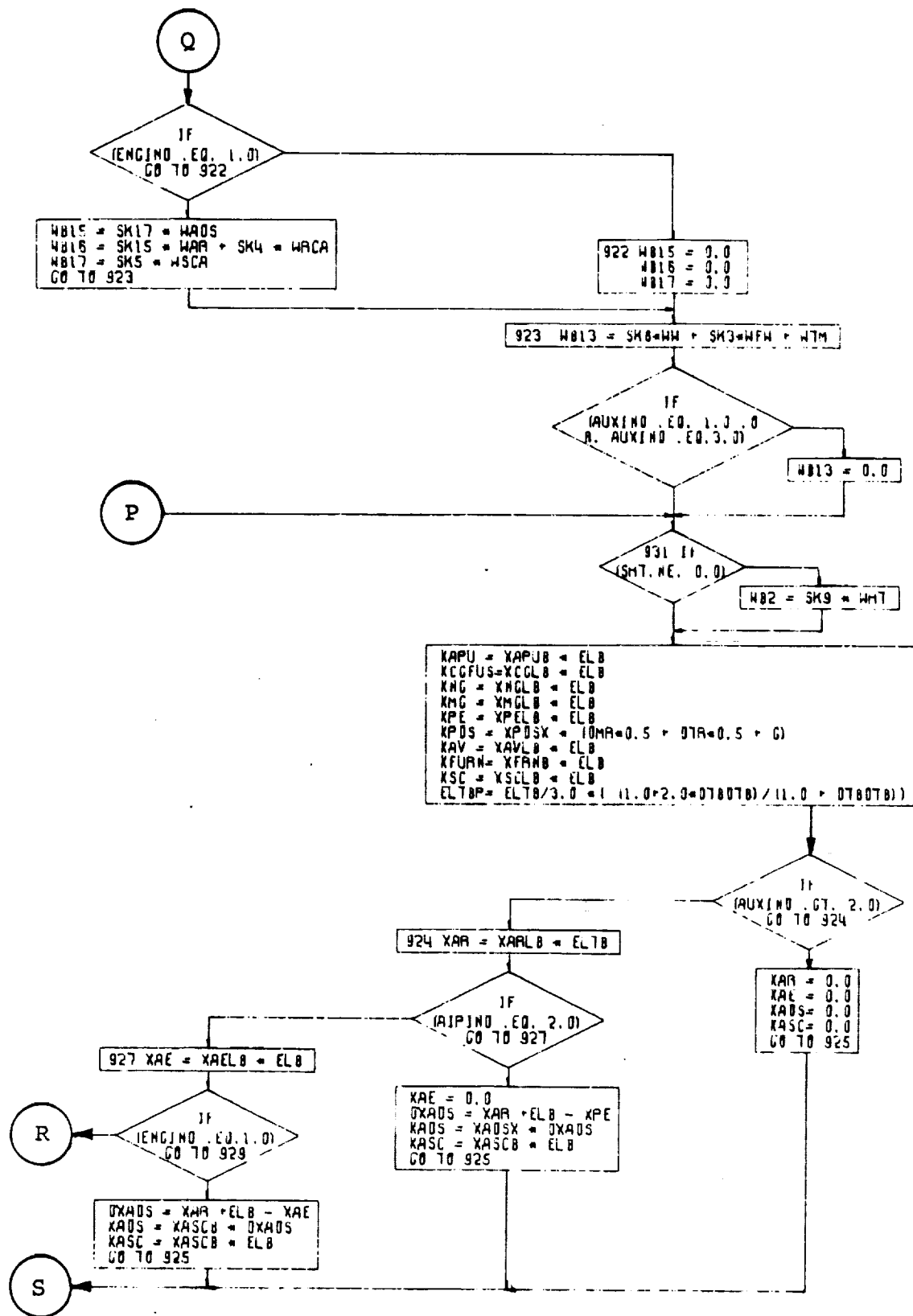


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 9 of 13).

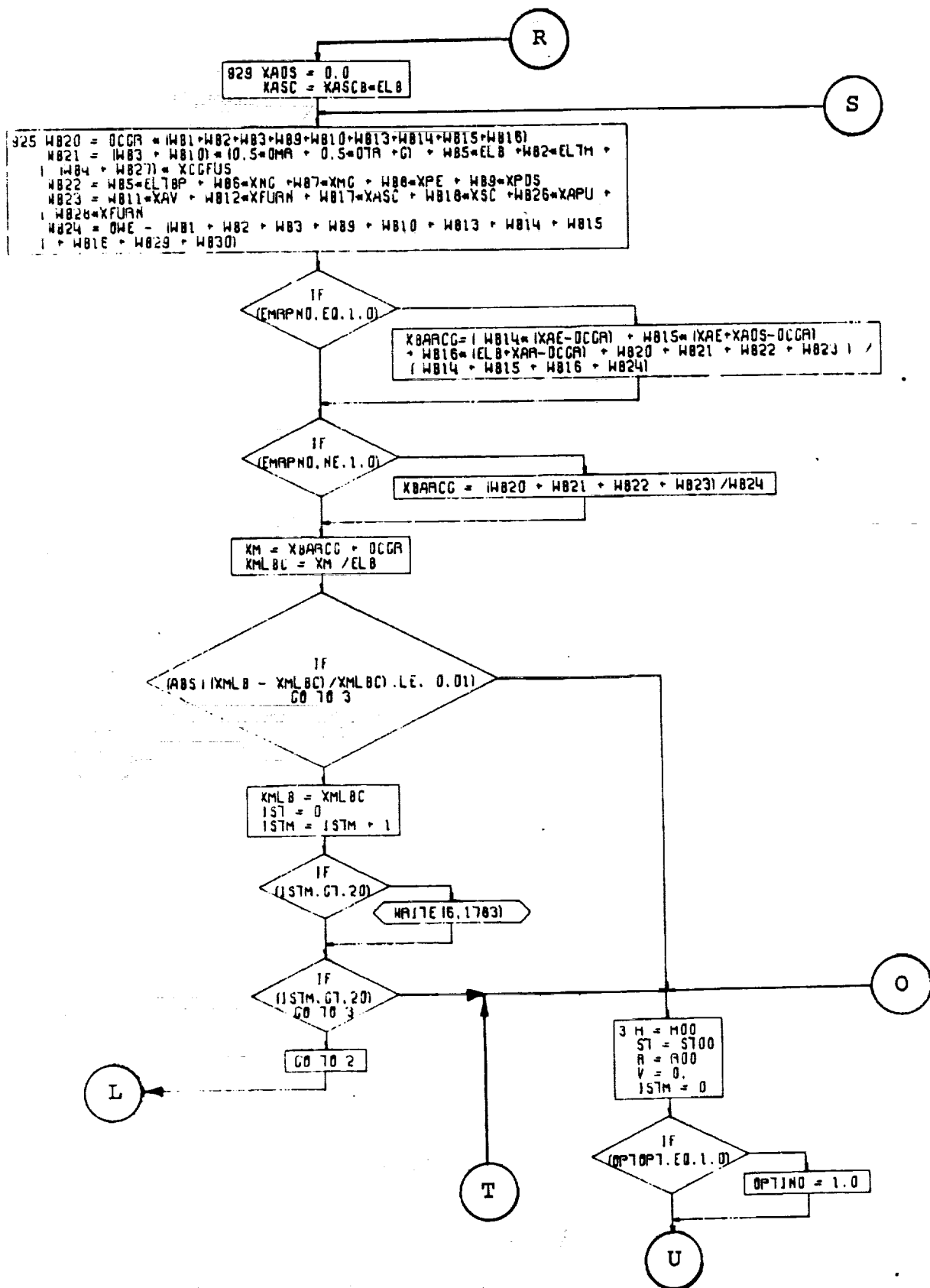


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 10 of 13).

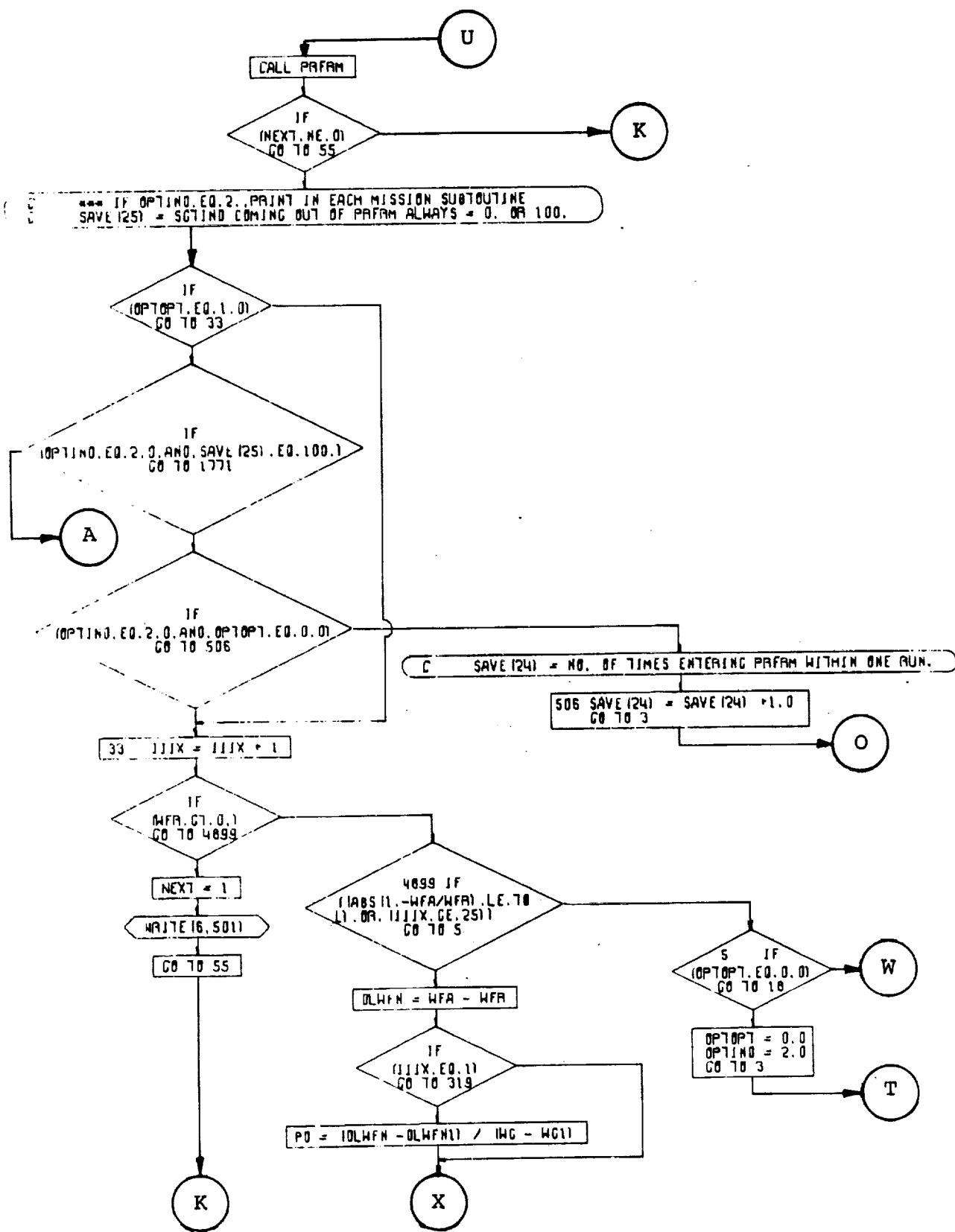


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 11 of 13).

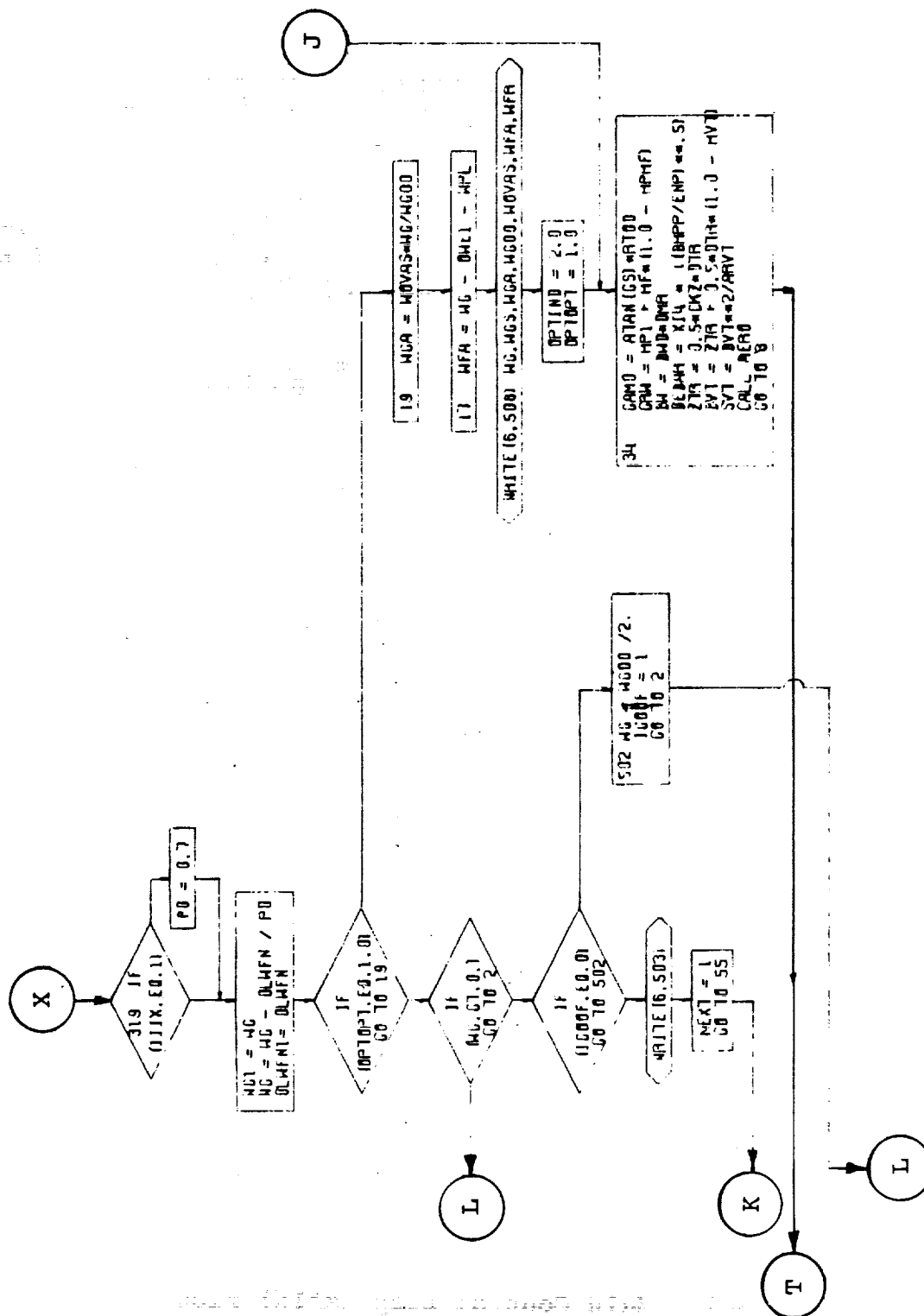


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 12 of 13).

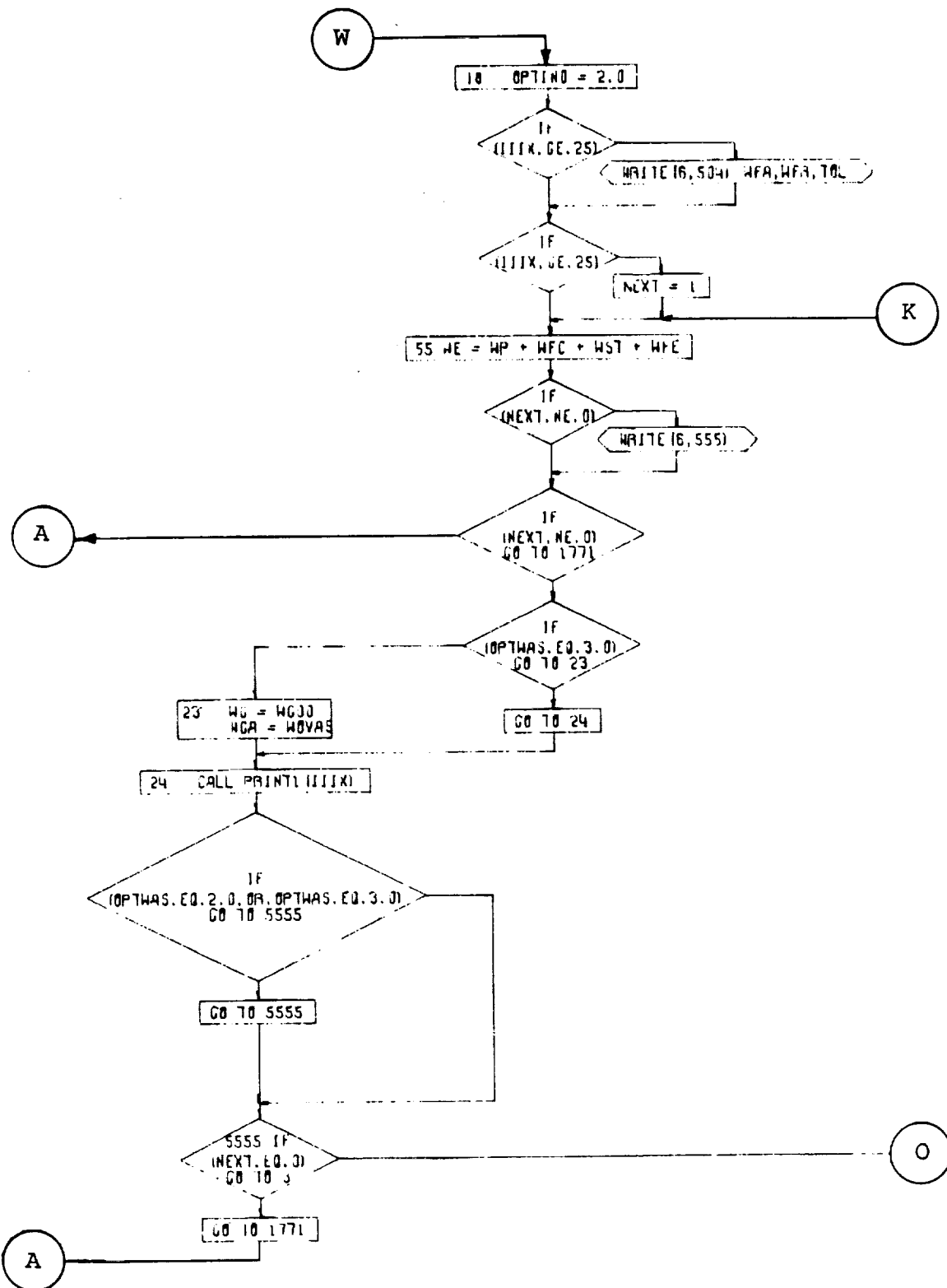


Figure 4-1. Main Control Loop (MAIN) Flow Chart
(Part 13 of 13).

4.2 ATMOSPHERE SUBROUTINE

The atmosphere subroutine will calculate the atmospheric density, pressure, and temperature as a function of altitude. Three options included below are available. These are specified by means of an input indicator, ATMIND, which is input individually for the performance data and the engine sizing data. Thus, the atmosphere can be calculated differently for each segment of the flight profile and for the engine sizing.

The options are:

ATMIND = 0: Standard atmosphere

ATMIND = 1: Constant increment in temperature above standard temperature

ATMIND = 2: Nonstandard temperature distribution as a function of altitude

The flow chart for the atmosphere subroutine is shown in Figure 4-2.

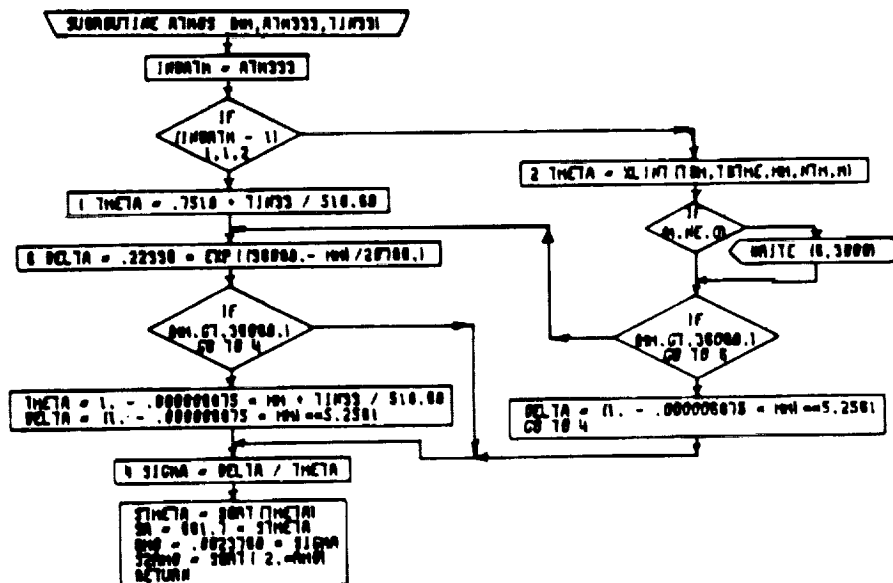


Figure 4-2. Atmosphere Subroutine, Flow Chart.

4.3 DRAW CALCULATIONS SUBROUTINE

The drag calculations subroutine uses the factors a_5 through a_9 , as determined by the aerodynamics calculations subroutine to calculate the total drag of the helicopter. Besides parasite drag, in the case of compound or winged helicopters, total drag includes wing induced drag and rotor/wing interference drag, the latter being calculated using a simplified Prandtl Bi-Plane Theory approach. The total helicopter propulsive thrust coefficient (C_X) is calculated as a function of forward flight helicopter advance ratio (μ). The subroutine flow chart is shown in Figure 4-3.

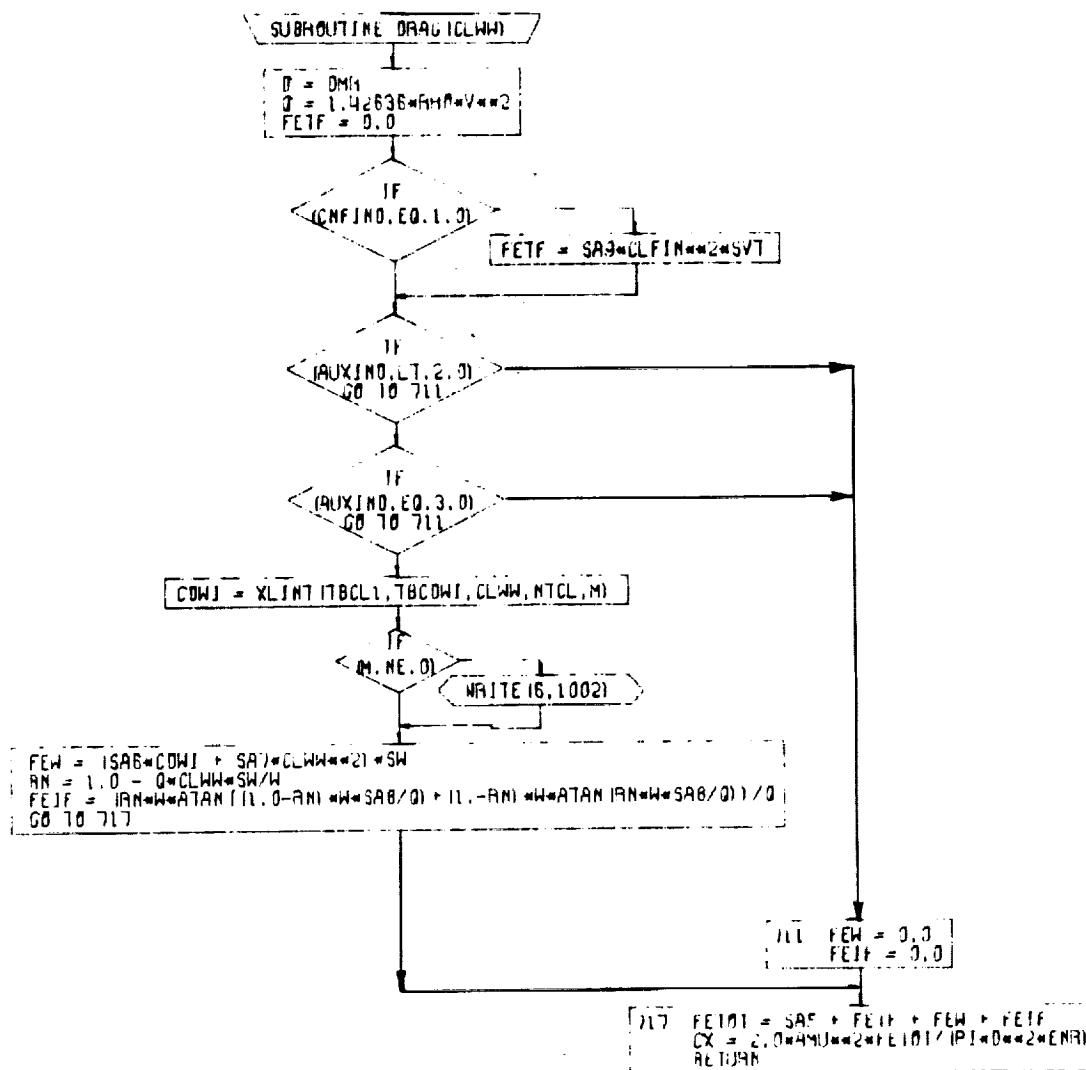


Figure 4-3. Drag Calculations (DRAG) Subroutine Flow Chart (Part 1 of 1).

4.4 ENGINE LIBRARY AND ENGINE CYCLE SUBROUTINES

The basic cycle performance data consists of tabulated values of four variables: thrust (power), fuel flow, gas generator shaft rpm, and power turbine shaft rpm. For the primary engine cycles, these tables are functions of Mach number and turbine inlet temperature. For lift engine cycles, the tables are functions only of turbine inlet temperature. All data are in referred, normalized format as shown in Table 4-1.

TABLE 4-1
ENGINE CYCLE DATA FORMAT

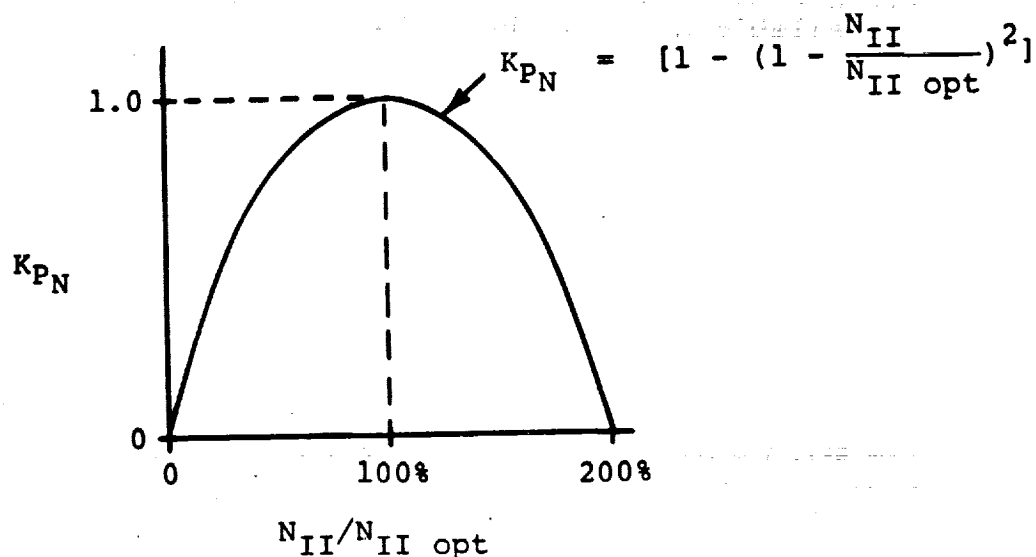
VARIABLE	SYMBOL	REFERRED, NORMALIZED FORM
Thrust	F_N	$F_N/\delta F_N^*$
Power	SHP	$SHP/\delta \sqrt{\theta} SHP^*$
Gas Generator rpm	N_I	$N_I/\sqrt{\theta} N_I^*$
Power Turbine rpm	N_{II}	$N_{II}/\sqrt{\theta} N_{II}^*$
Fuel Flow	$\dot{W}_f$	$\begin{cases} \dot{W}_f/\delta \sqrt{\theta} F_N^* \\ \dot{W}_f/\delta \sqrt{\theta} SHP^* \end{cases}$
Turbine Inlet Temperature	T	T/θ
Where:	* = Max. Power Setting, Static, Sea Level, Standard Day θ = Ambient Temperature ($^{\circ}R$) Divided by 518.69$^{\circ}R$ δ = Ambient Pressure (psia) Divided by 14.696 psia	

The standard engine cycle library consists of forty-five different generalized engine cycles shown in Table 4-2. The data for each cycle is punched in card form, accessible for input with the remainder of the input data for a given case. Each cycle is numbered; and, to guard against selection of an incorrect cycle, the cycle number is checked against a similar number input to the program by the user.

TABLE 4-2						
HESCOMP ENGINE LIBRARY						
		Engine Cycle Number	Maximum Turbine Inlet Tempera- ture - °R	Compressor Design Pressure Ratio	Fan Bypass Ratio	
↑ Auxiliary Independent Propulsion Engines ↓	↑ Primary Propulsion Engines ↓		1	2600	13	
			2	2600	16	
		Turboshaft Engines	3	2900	13	
			4	2900	16	
			5	2900	19	
			6	3200	13	
			7	3200	16	
			8	3200	19	
			9	3200	22	
	Turbojet Engines		10	2600	13	
			11	2600	16	
			12	2900	13	
			13	2900	16	
			14	2900	19	
			15	3200	13	
			16	3200	16	
			17	3200	19	
			18	3200	22	
	Turbofan Engines		19,20,21	2600	16	2,4,6
			22,23,24	2600	20	2,4,6
			25,26,27	2900	16	2,4,6
			28,29,30	2900	20	2,4,6
			31,32,33	2900	24	2,4,6
			34,35,36	3200	16	2,4,6
			37,38,39	3200	20	2,4,6
			40,41,42	3200	24	2,4,6
			43,44,45	3200	28	2,4,6

The fuel flow of the basic engine cycle should correspond to the manufacturer's specification data. Adjustments to the fuel flow level may be made by means of the input multiplier, K_{FF} .

Because of the normalized, referred format, all data are valid for any ambient temperature, standard or nonstandard. With the exception of referred power, none of the tables are dependent upon power turbine speed. By setting $N2IND = 2$, turbo-shaft engine power at nonoptimum N_{II} will be calculated by the program by multiplying power at optimum N_{II} by a correction factor, K_{PN} , which is a function of $N_{II}/N_{II\text{ OPT}}$. The factor K_{PN} is normally calculated by the program and obeys a second-order relationship:

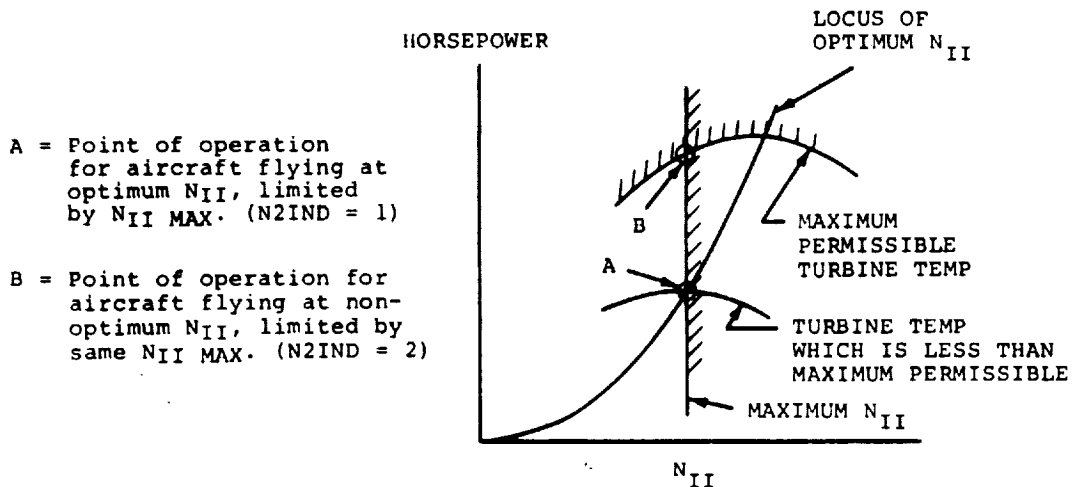


Most, but not all, turboshaft engines will obey this relationship. For engine cycles whose performance is not properly represented by the above curve, the user may input a table of K_{PN} versus $N_{II}/N_{II\text{ OPT}}$. The program user inputs $N_{II}/N_{II\text{ MAX}}$ for each flight segment and $N_{II\text{ MAX}}/N_{II}^*$ for the engine cycle. The program uses this information to establish the value of $N_{II}/N_{II\text{ OPT}}$ for each point of flight.

By setting $N2IND = 0$ or 1, the program will assume that the power turbine is always operating at optimum speed and no correction will be applied. $N2IND = 0$ will simulate an engine cycle which is operating at optimum N_{II} and for which no upper limit has been placed on N_{II} . For many applications, this option will be perfectly adequate for preliminary sizing studies. The adequacy of this assumption can be determined by consideration of the following factors:

1. It may be desirable; e.g. as in the case of a slowed-rotor compound helicopter, to reduce the main rotor rpm in cruise flight.
2. For some applications this may, in turn, force the engine to operate at a very inefficient N_{II} . In general, the optimum N_{II} increases as output power increases relative to the maximum level.

$N2IND = 1$ will simulate operation of an engine cycle at optimum N_{II} , but with the restriction of a maximum value for N_{II} . This type of operation is characteristic of airplanes employing fixed pitch propellers. Care should be taken in using this option because it may lead to a significant reduction in power available as shown by the sketch below:



Limitations on engine cycle operation may be input to the program on any combination of the following: fuel flow, torque, gas generator speed, gas generator referred rpm, or output shaft speed. Engine ratings (power settings) are dictated by turbine temperature. Five discrete values of that parameter are input for the primary engine cycles, one for each of the following power settings: maximum, military, normal, flight idle, and ground idle.

The program will print out, during the mission, the value of turbine temperature and a code that designates which condition is governing the engine performance at that point: power or thrust required, turbine temperature, torque limit, N_I limit, referred N_I limit, N_{II} limit, or fuel flow limit.

Manufacturer's data on some engines show significant variations in both referred power ($\text{shp}/\delta\sqrt{\theta}$) and lapse rate with respect to changes in altitude. These variations are due to Reynolds' number effects. It has been found that these effects can be accounted for by means of a multiplicative factor on power available which is a function of the Reynolds number based on compressor inlet conditions, compressor blade geometry, and tip speed. Figure 4-4 shows a typical curve for a real engine. The correction factor K_{PR} is input to the program as a function of the Reynolds' parameter

$$\frac{N_I}{N_I^*} \frac{D}{v_1} .$$

The tabular input of power, fuel flow, N_I , and N_{II} for engines which require Reynolds number corrections should be input to the program at a nominal fixed value of the Reynolds number parameter. The K_{PR} correction factor will then give the power at other values of the Reynolds number parameter. In the example shown in Figure 4-4, the nominal value of the parameter was chosen as 9000 seconds/foot.

The referred N_I limit is a constraint on the value of $N_I/\sqrt{\theta_1}$ where θ_1 is the temperature ratio at the compressor face. This limit simulates a restriction on compressor speed. The user inputs a maximum value of $N_I/N_I^*\sqrt{\theta_1}$.

The engine dry weight and dimensions are calculated by means of the input parameters k_3 , k_{3I} , k_4 , k_{4I} , ξ_4 and ξ_{4I} :

$$\text{Primary engines} \left\{ \begin{array}{l} \text{weight (lb)} = k_3 \frac{\text{SHP}^*}{N_P} + k_4 \\ \text{diameter (ft)} = \xi_4 \left[\frac{\text{SHP}^*}{N_P} \right]^{1/2} \\ N_P = \text{number of primary engines} \end{array} \right.$$

$$\text{Auxiliary Independent Engines} \left\{ \begin{array}{l} \text{weight (lb)} = k_{3I} \frac{F_N^*}{N_{Pi}} + k_{4I} \text{ or } k_{3I} \frac{\text{SHP}^*_i}{N_{Pi}} + k_{4I} \\ \text{diameter (ft)} = \xi_{4I} \left[\frac{F_N^*}{N_{Pi}} \right]^{1/2} \text{ or } \xi_{4I} \left[\frac{\text{SHP}^*_i}{N_{Pi}} \right]^{1/2} \\ N_{Pi} = \text{number of independent auxiliary engines} \end{array} \right.$$

TURBOSHAFT ENGINE A

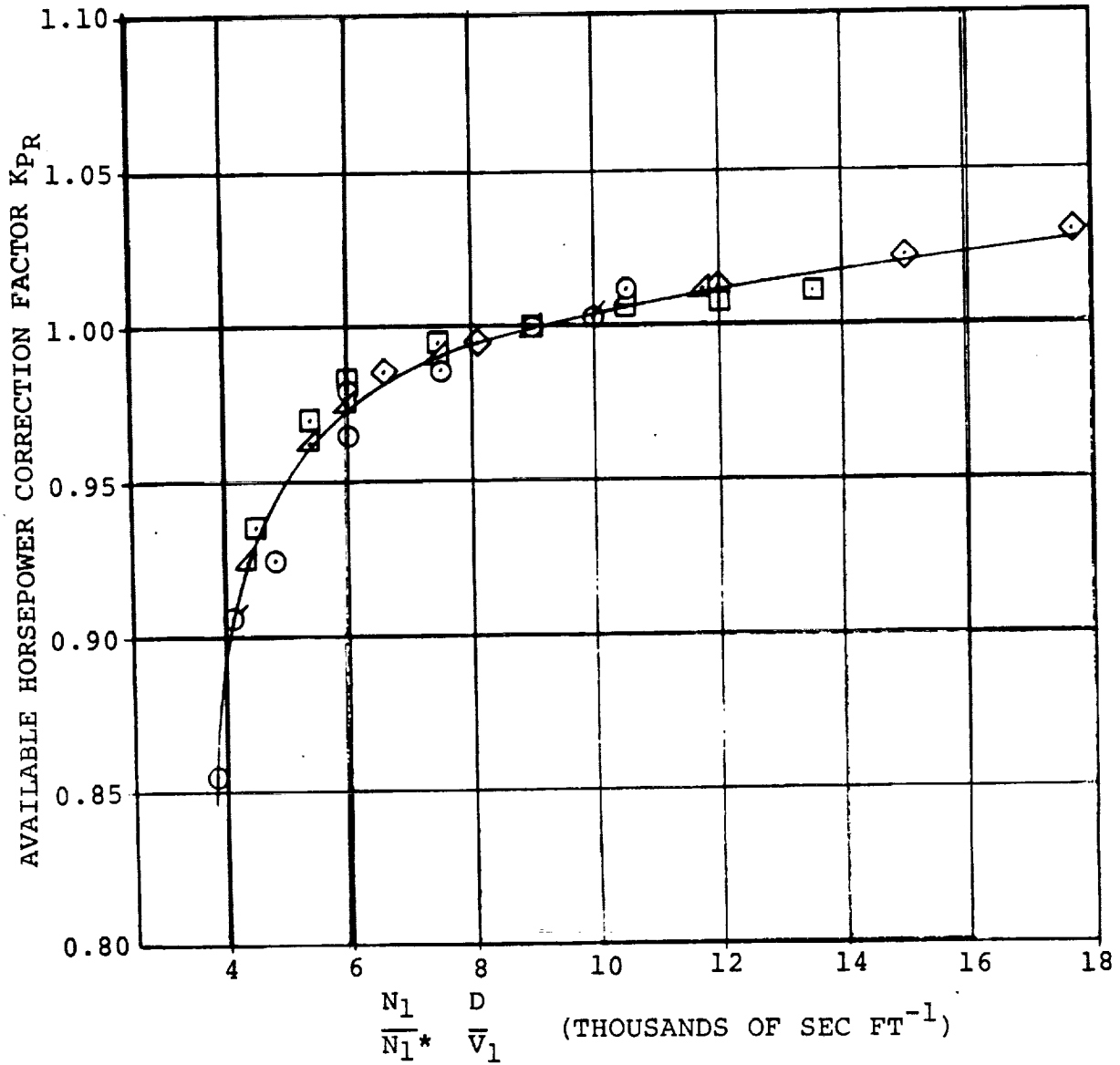


Figure 4-4. Typical Reynolds Number Correction Factor for a Turboshaft Engine Cycle.

It should be noted that auxiliary independent engine input data can be created from the engine cycle library data simply by the input of the applicable engine cycle IBM card deck, preceded and followed by a "66666" card. Nonstandard auxiliary independent engine performance is input using the sheet provided for that purpose.

Figures 4-5 through 4-12 are flow charts of the engine cycle subroutines. The purpose of these subroutines is described in Table 3-1 in Section 3-0 of this document.

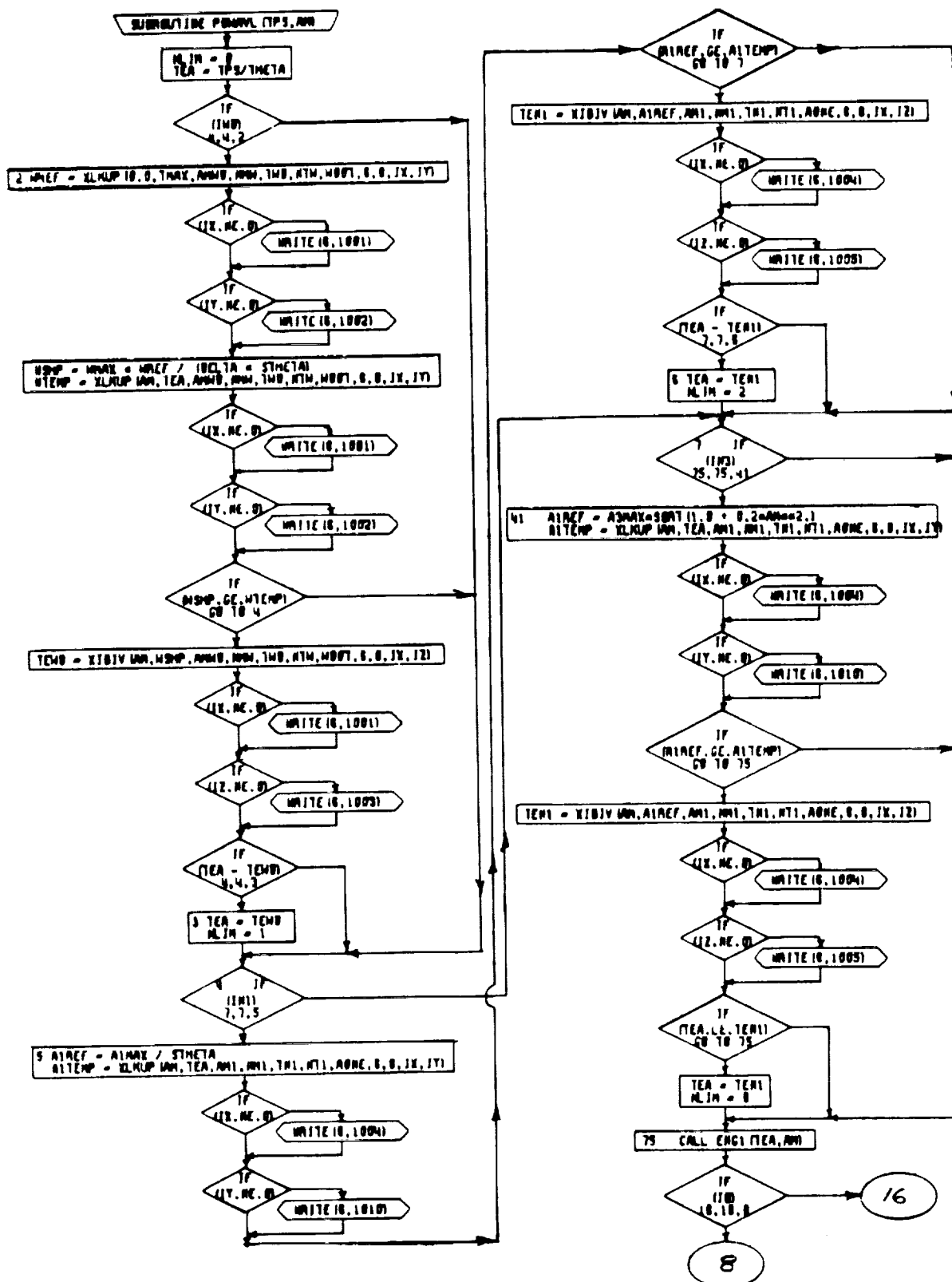


Figure 4-5. POWAVL Subroutine, Flow Chart (Part 1 of 3).

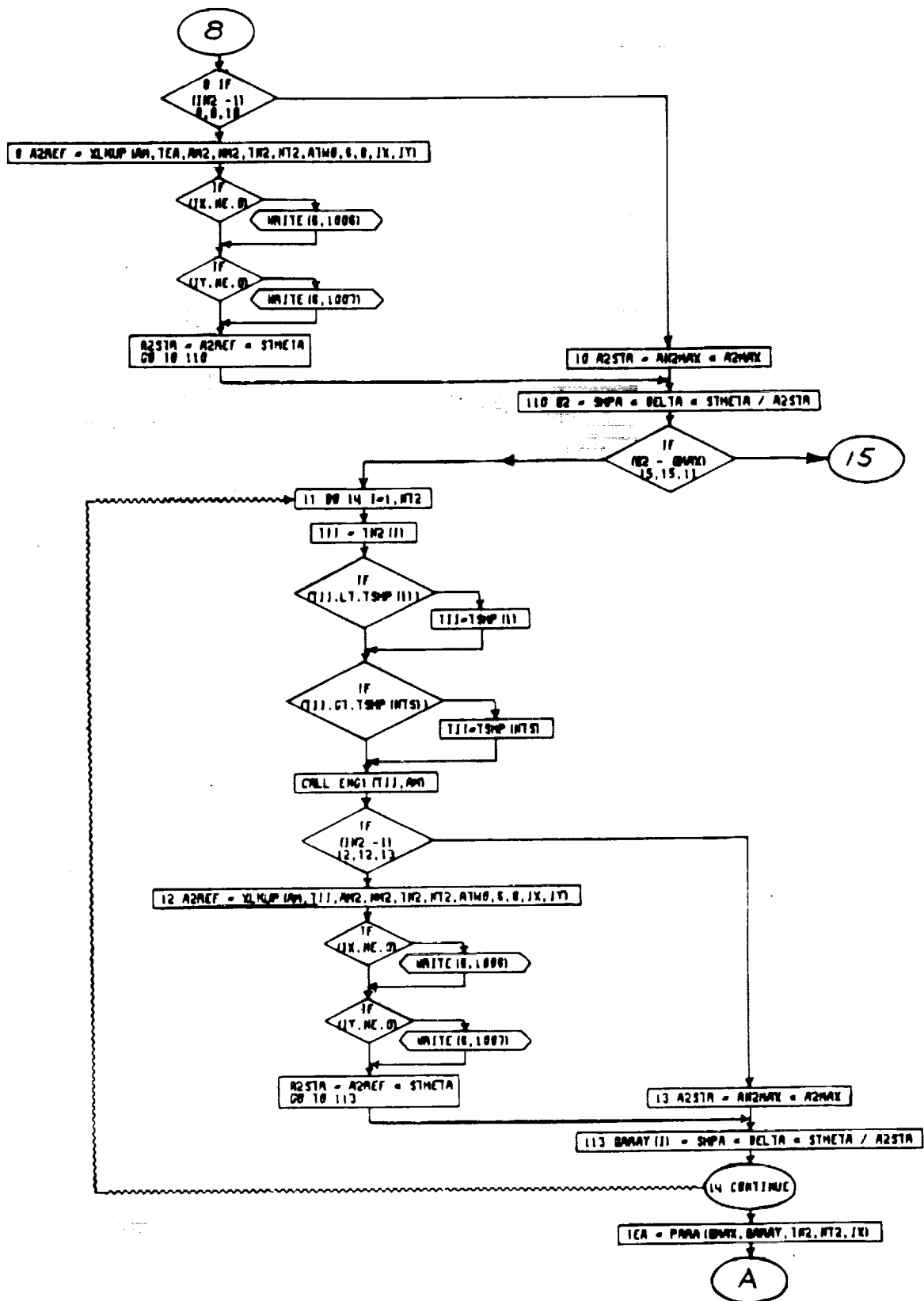


Figure 4-5. POWAVL Subroutine, Flow Chart (Part 2 of 3).

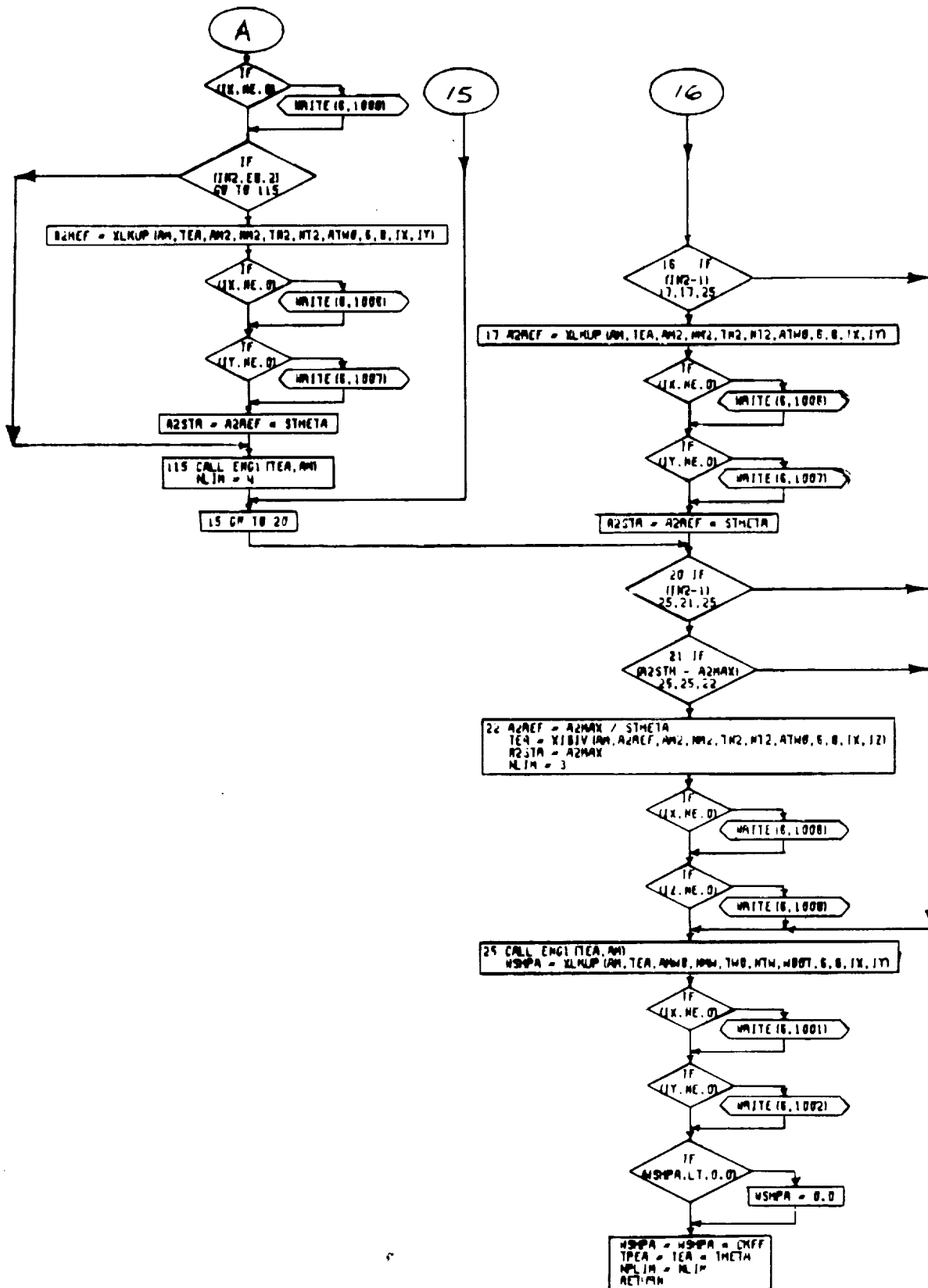


Figure 4-5. POWAVL Subroutine, Flow Chart (Part 3 of 3).

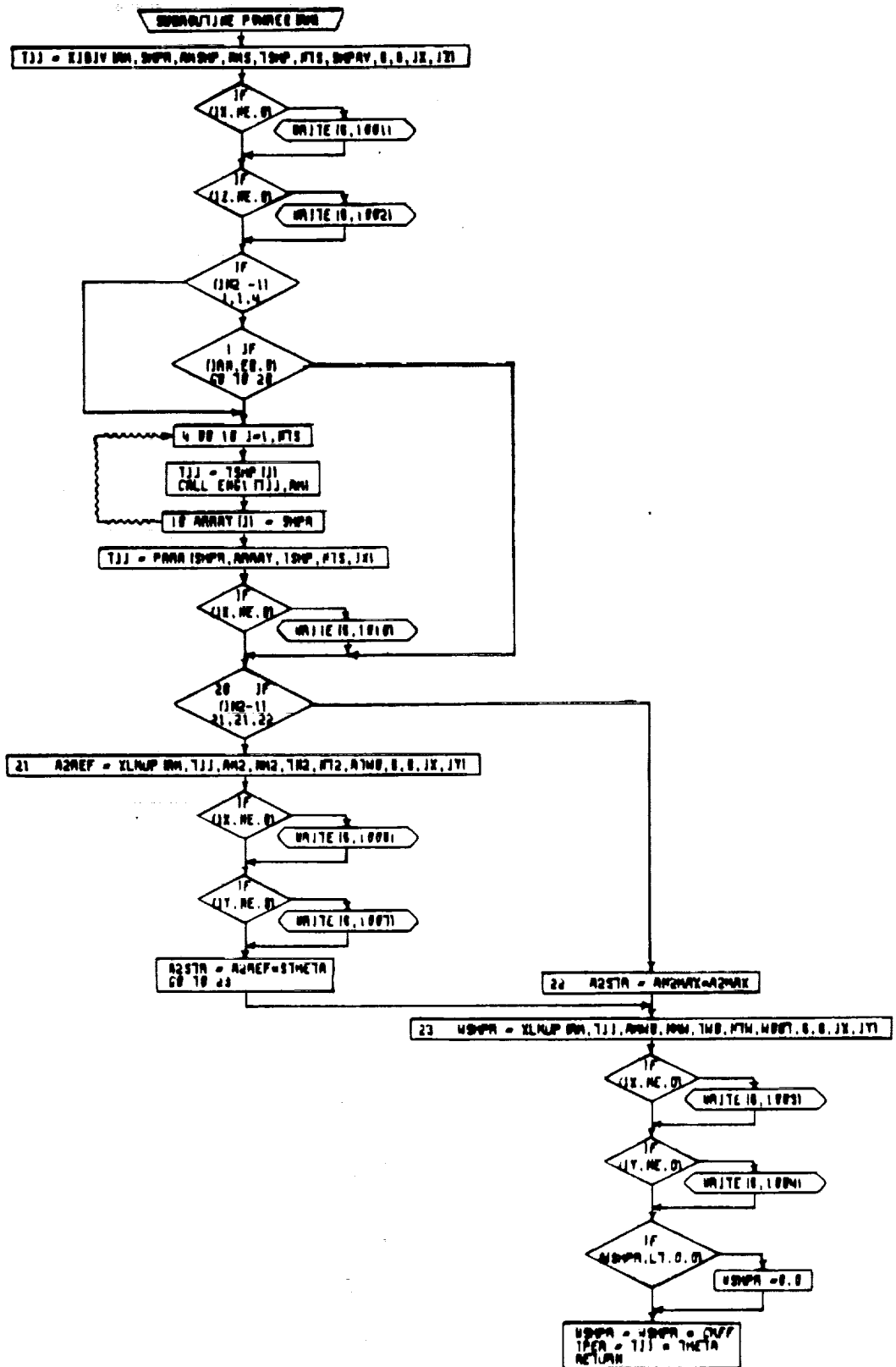


Figure 4-6. POWREQ Subroutine, Flow Chart.



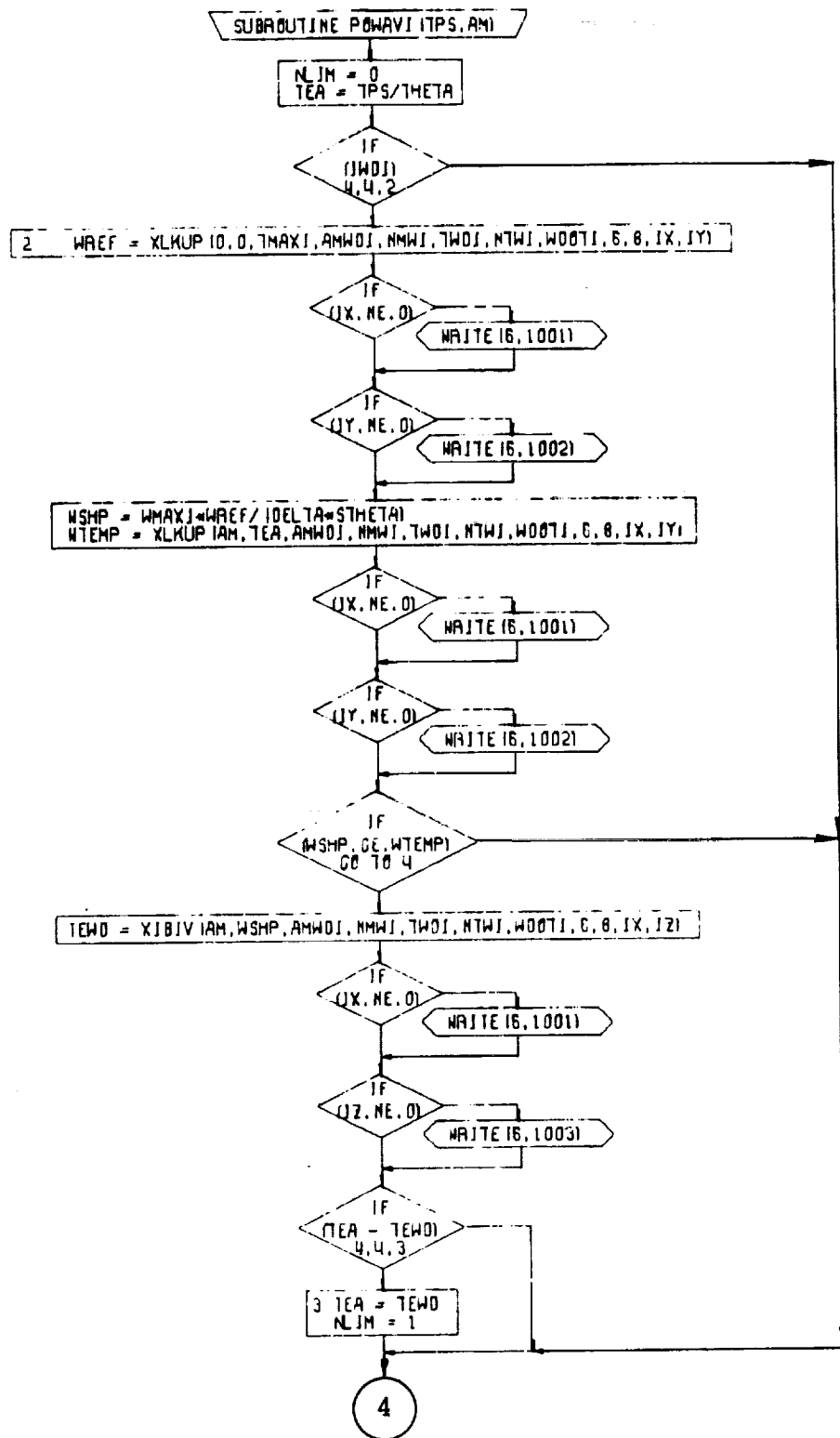


Figure 4-7. POWAVI Subroutine Flow Chart (Part 1 of 7).

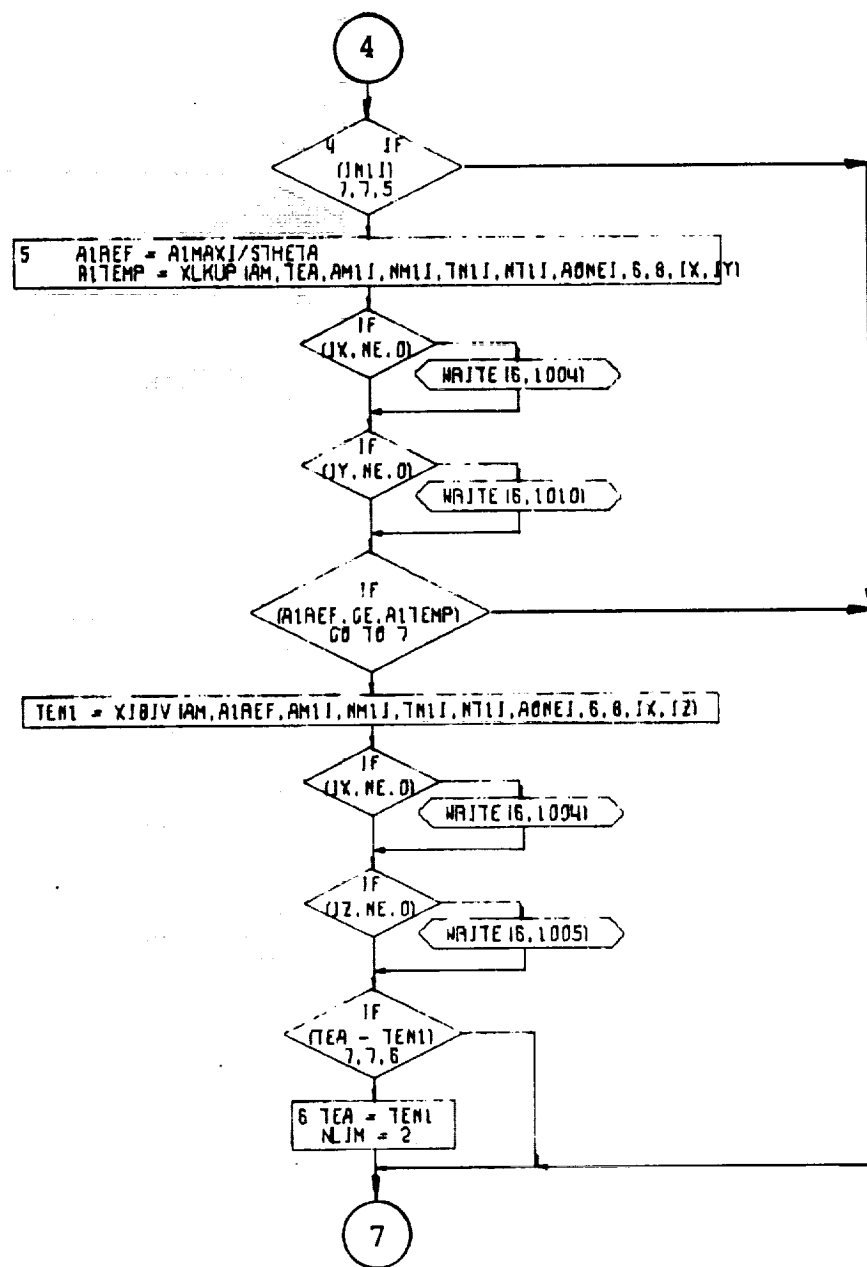


Figure 4-7. POWAVI Subroutine Flow Chart (Part 2 of 7).

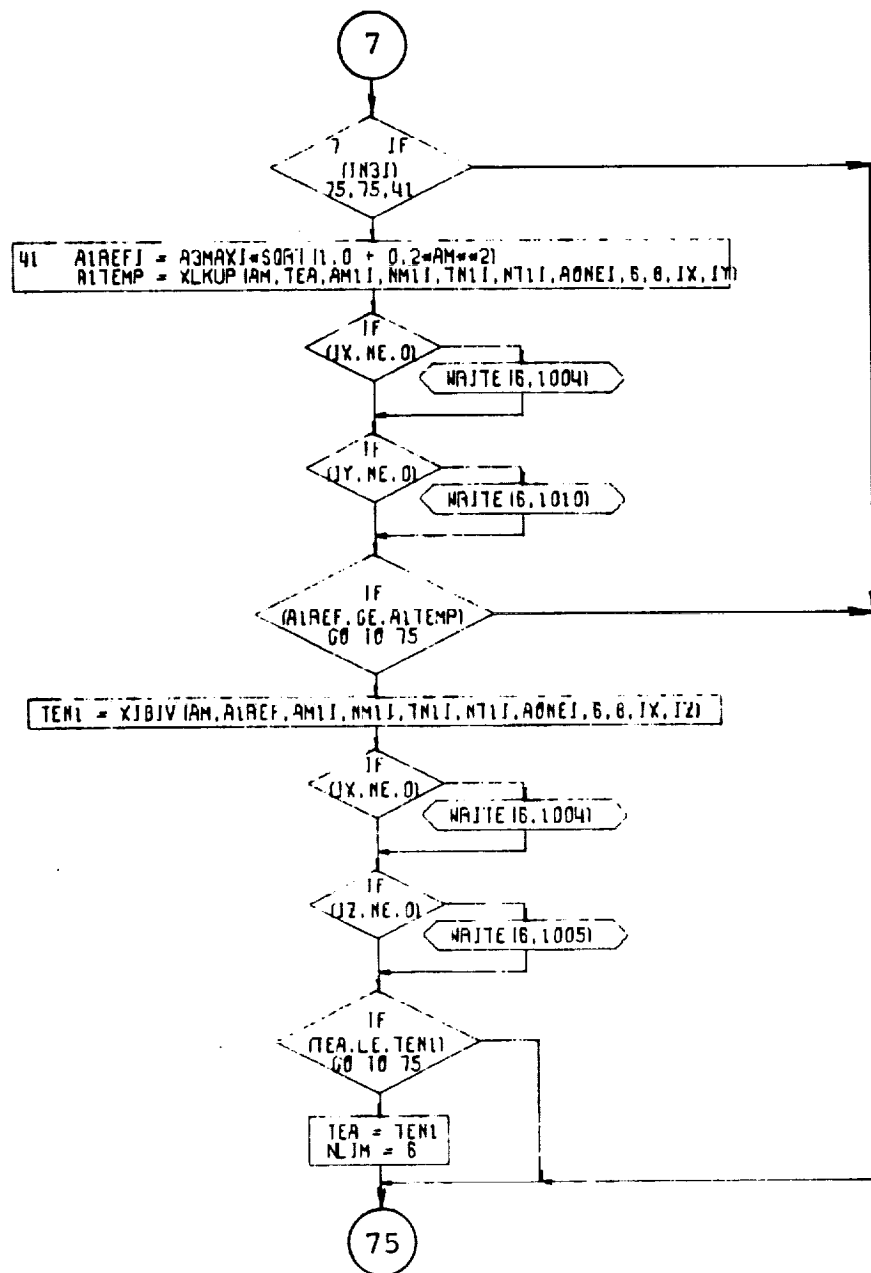


Figure 4-7. POWAVI Subroutine Flow Chart (Part 3 of 7).

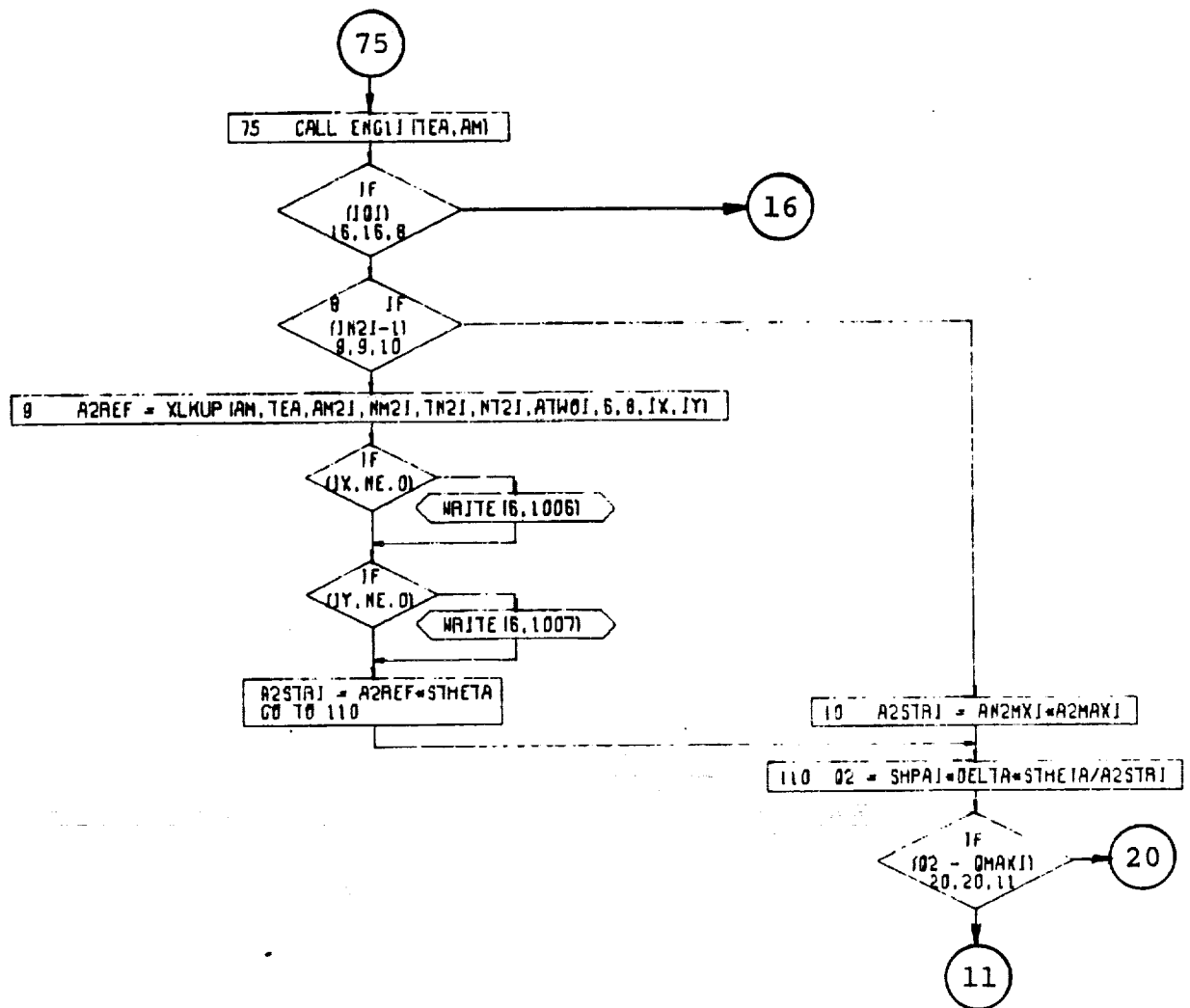


Figure 4-7. POWAVI Subroutine Flow Chart (Part 4 of 7).

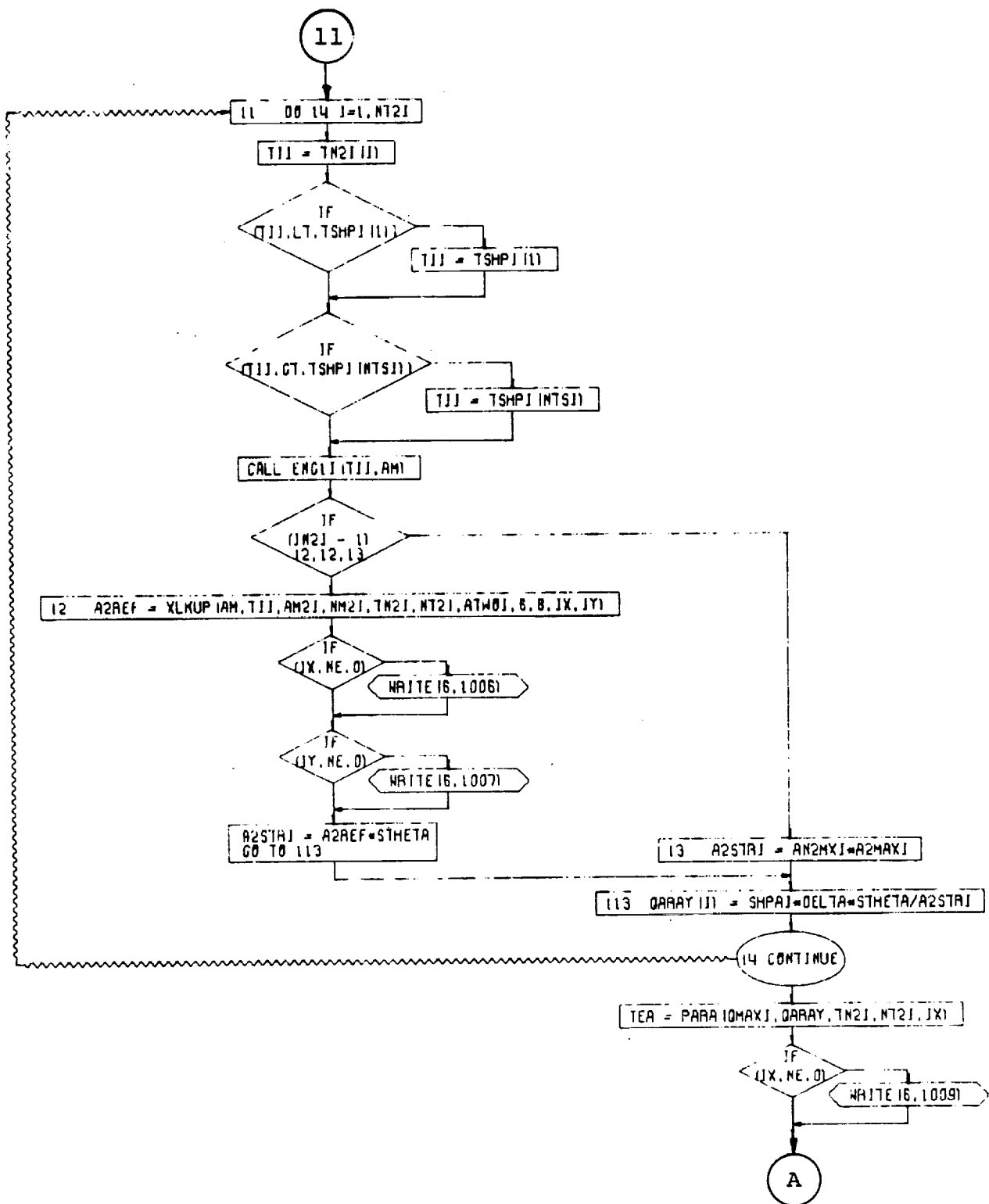


Figure 4-7. POWAVI Subroutine Flow Chart (Part 5 of 7).

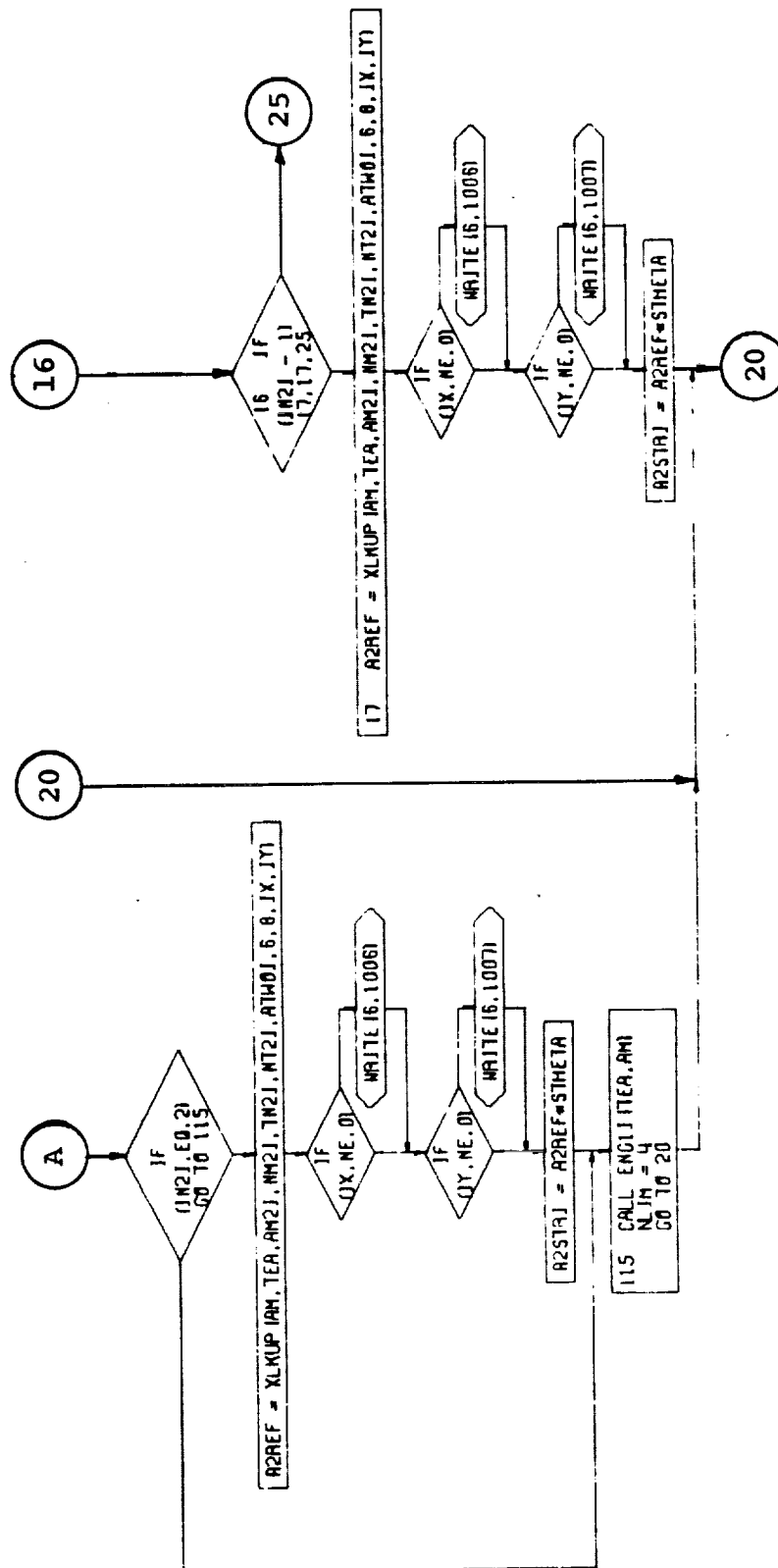


Figure 4-7. POWAVI Subroutine Flow Chart (Part 6 of 7).

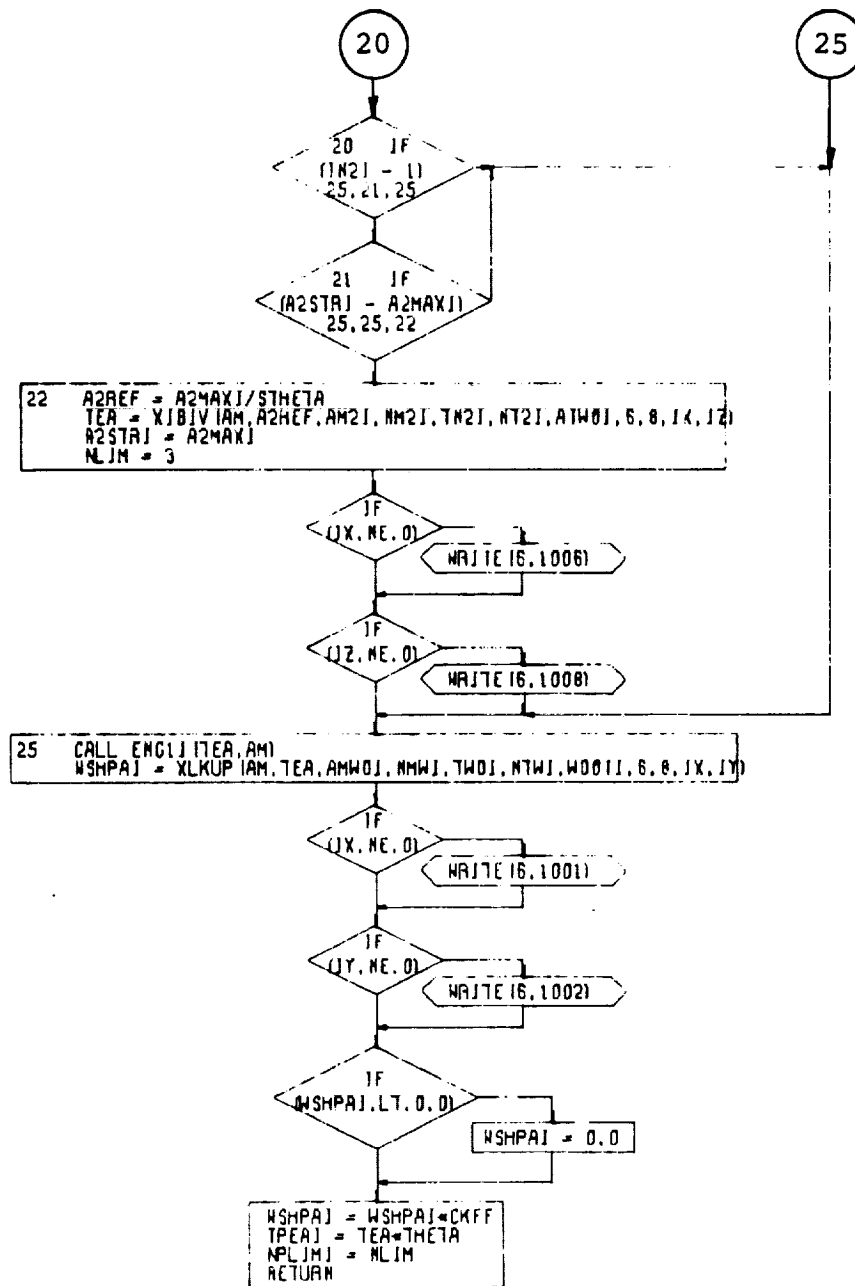


Figure 4-7. POWAVI Subroutine Flow Chart (Part 7 of 7).

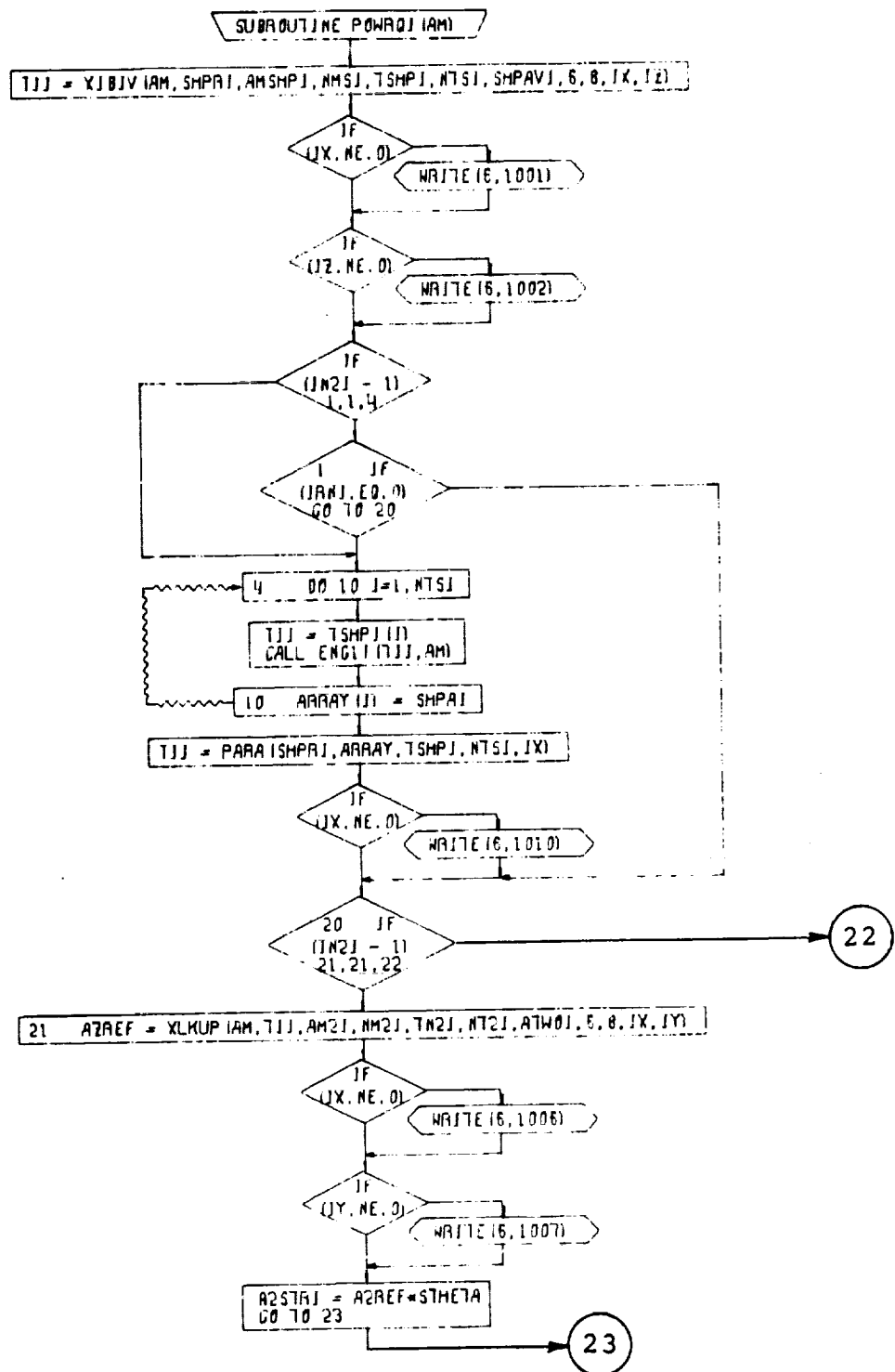


Figure 4-8. POWRQI Subroutine Flow Chart (Part 1 of 2).

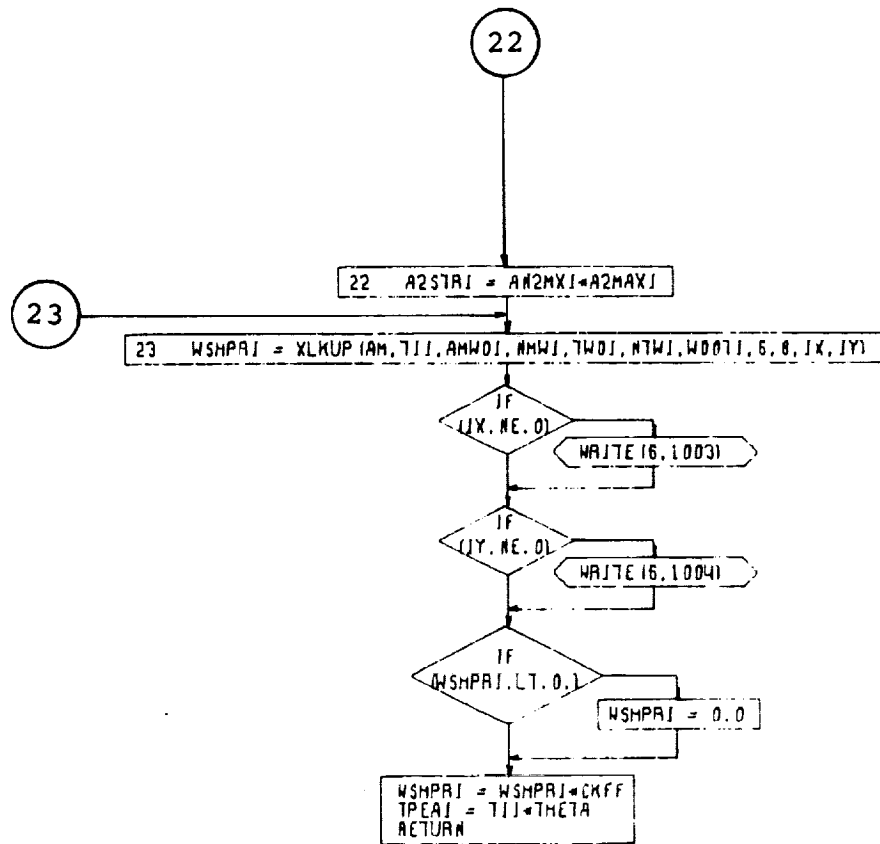


Figure 4-8. POWRQI Subroutine Flow Chart (Part 2 of 2).

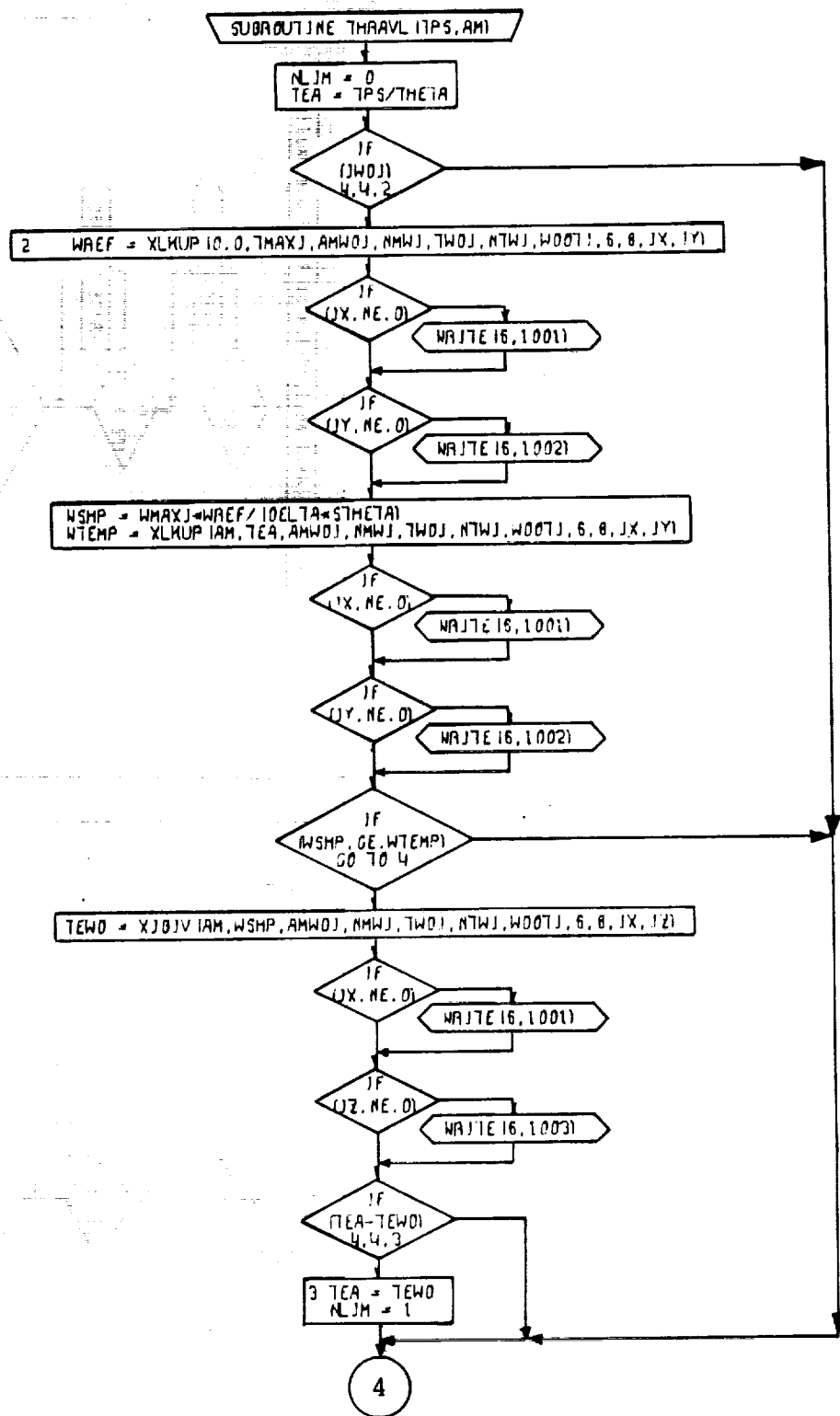


Figure 4-9. THR AVL Subroutine Flow Chart (Part 1 of 3).

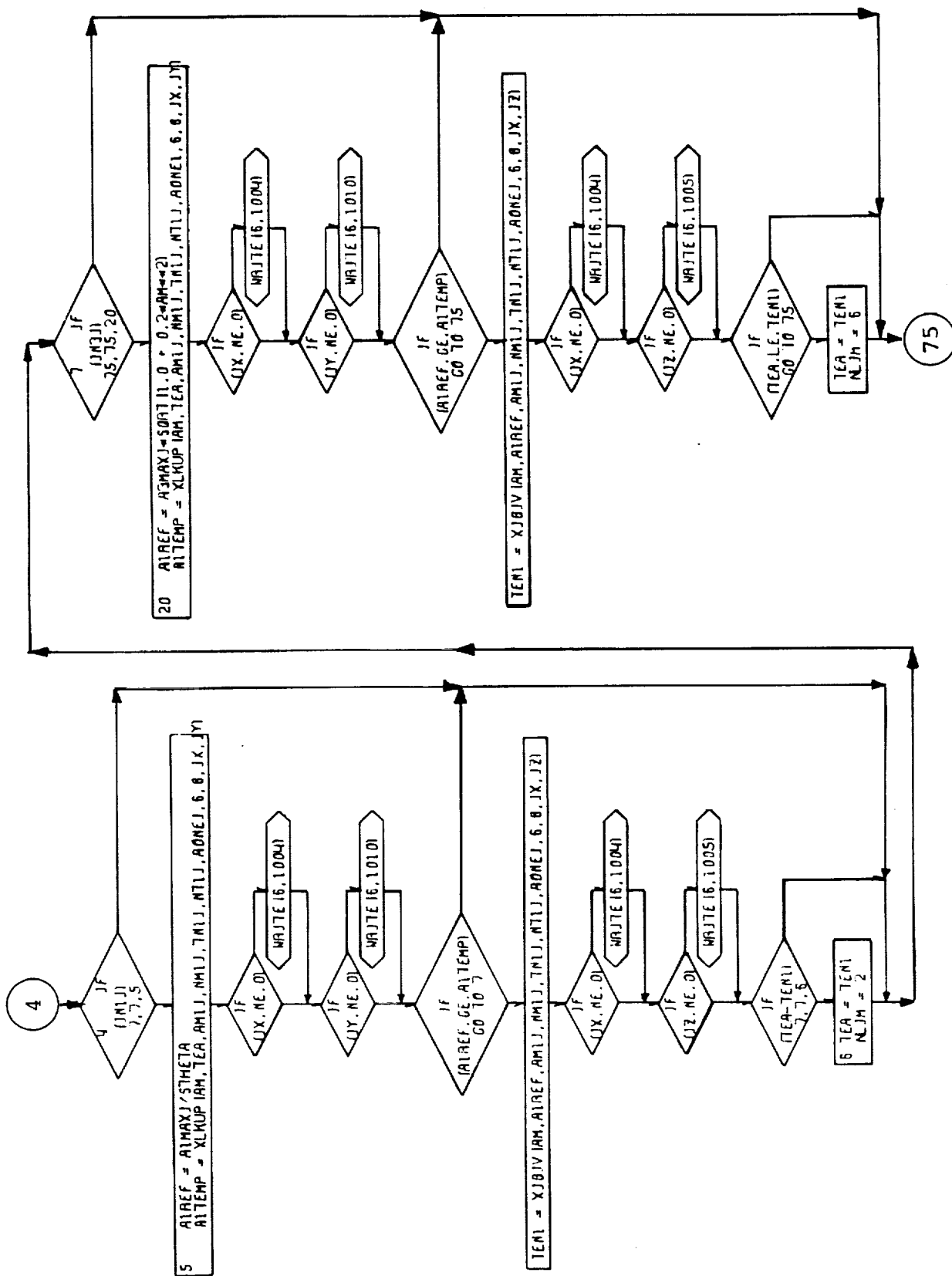


Figure 4-9. THRVL Subroutine Flow Chart (Part 2 of 3).

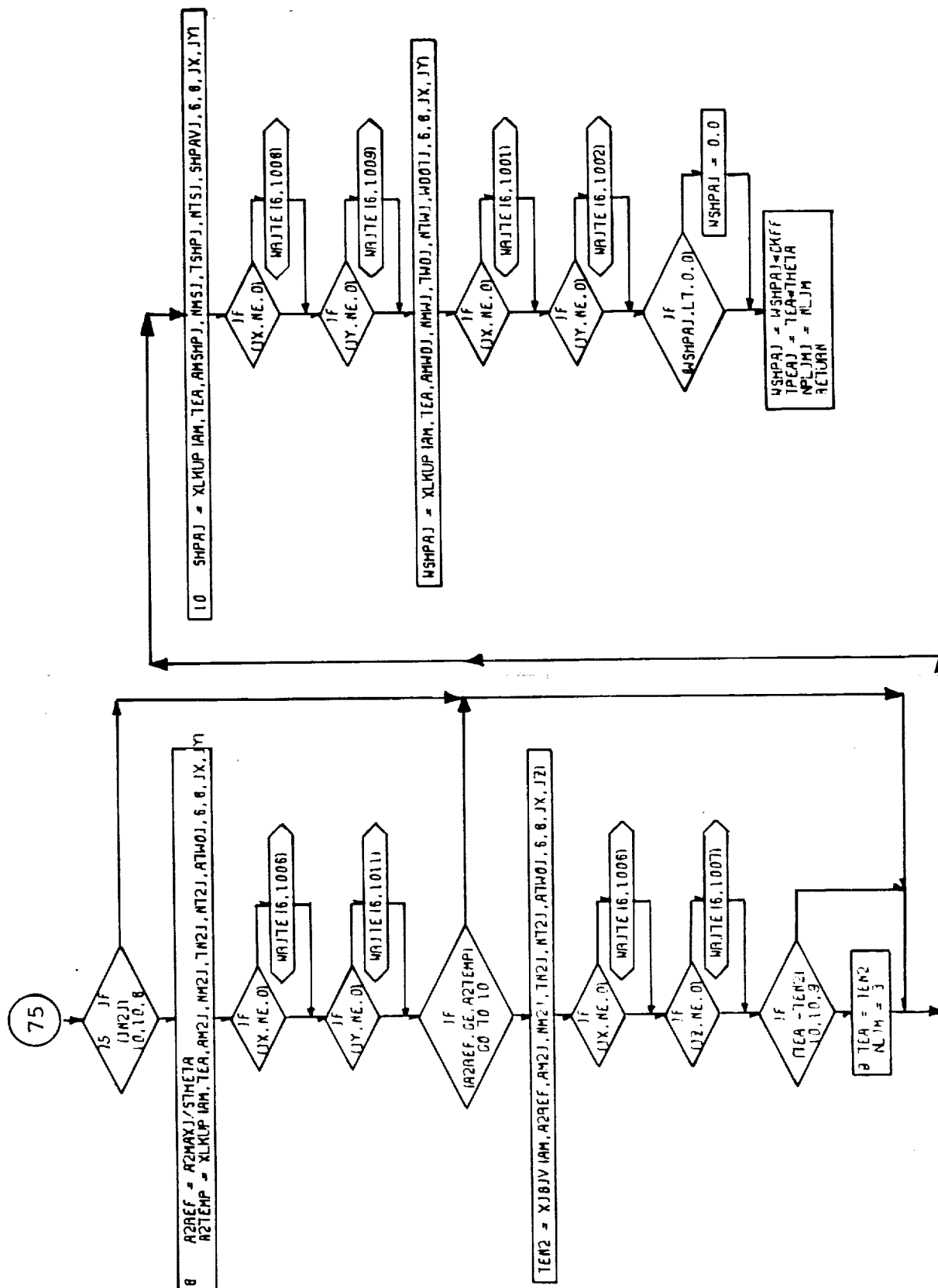


Figure 4-9. THRAVL Subroutine Flow Chart (Part 3 of 3) .

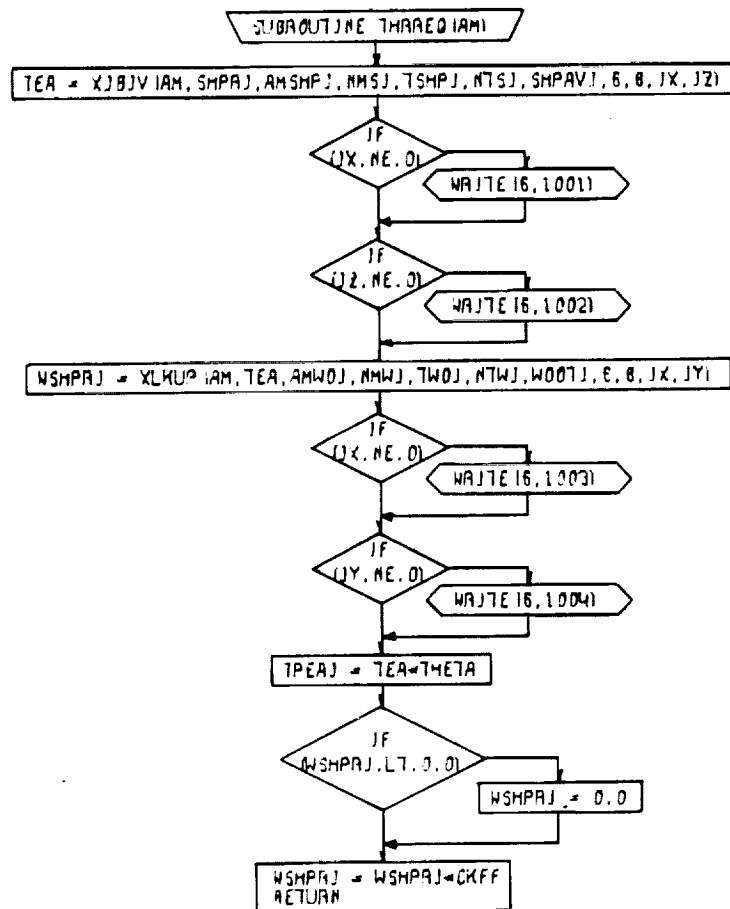


Figure 4-10. THRREQ Subroutine Flow Chart (Part 1 of 1).

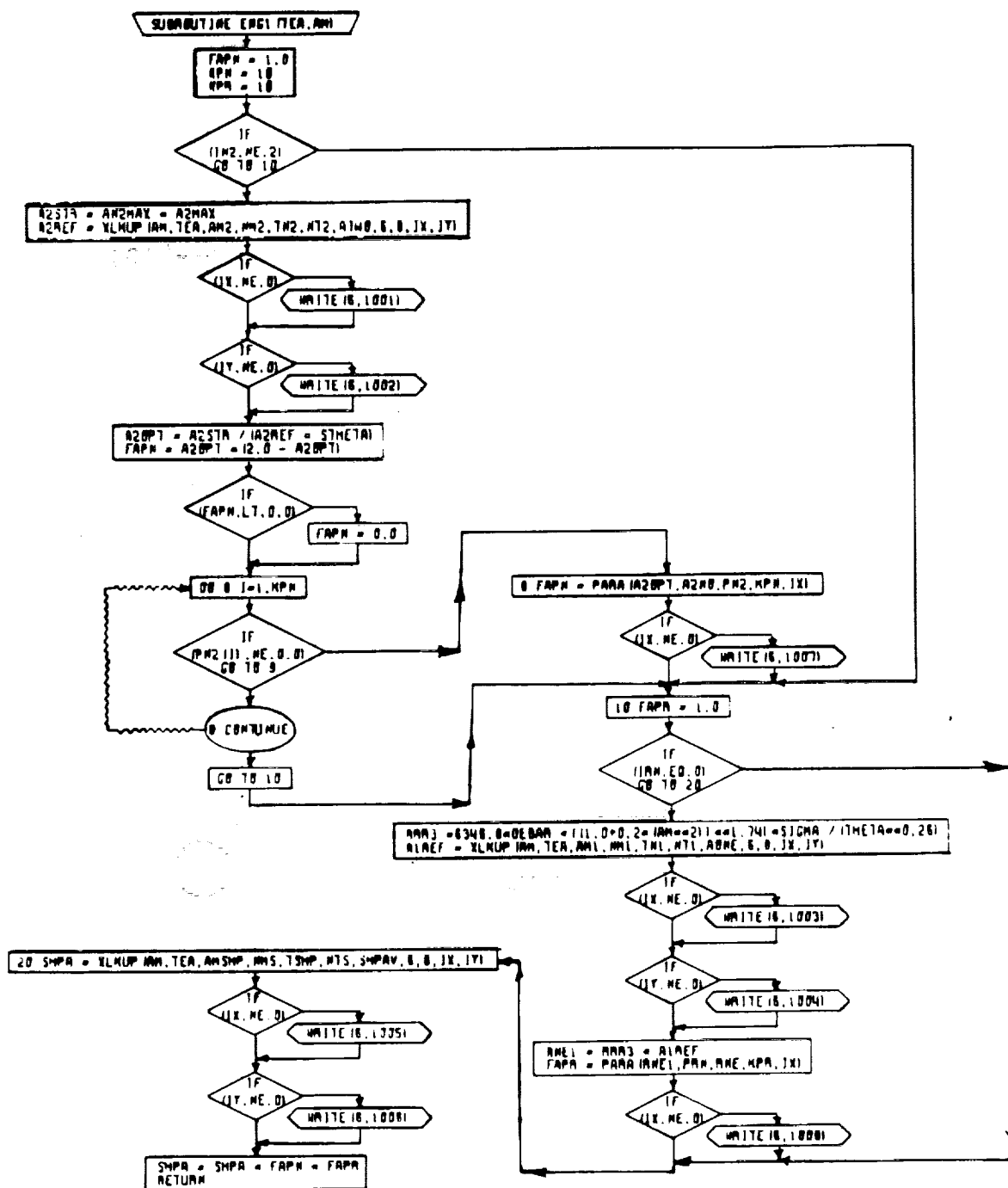


Figure 4-11. ENG 1 Subroutine, Flow Chart.

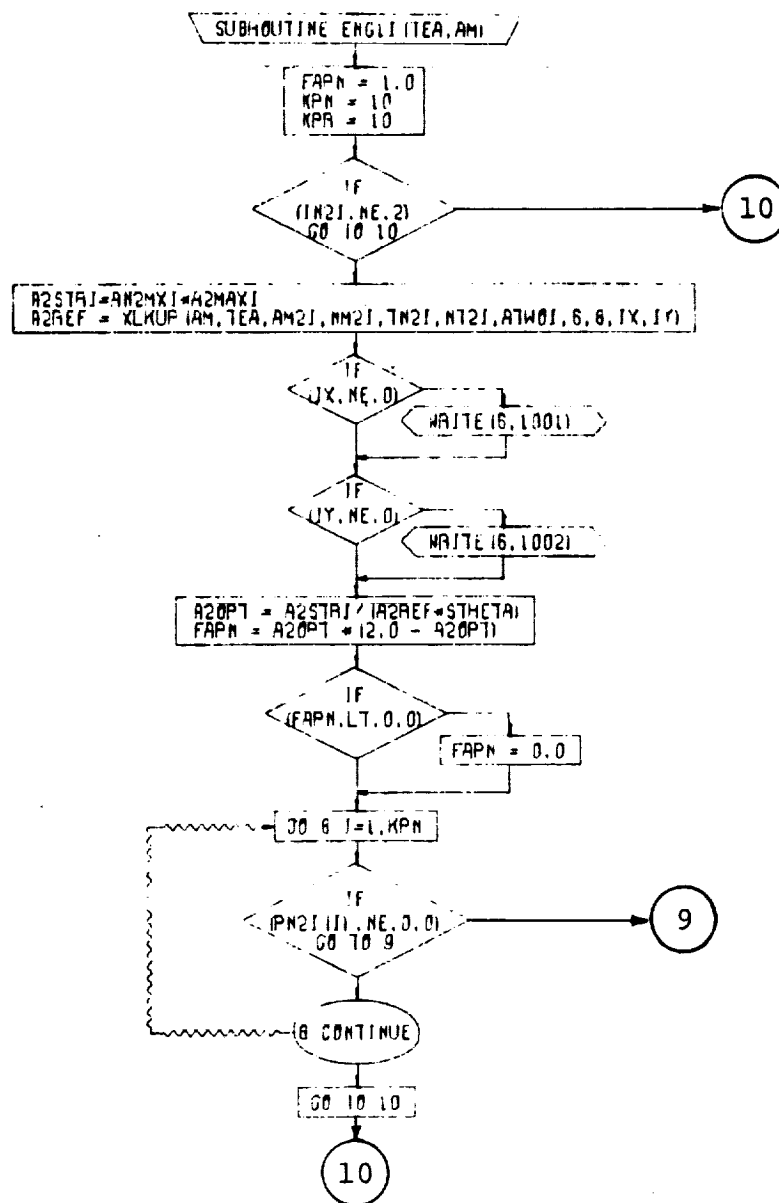


Figure 4-12. ENG 1 I Subroutine Flow Chart (Part 1 of 2).

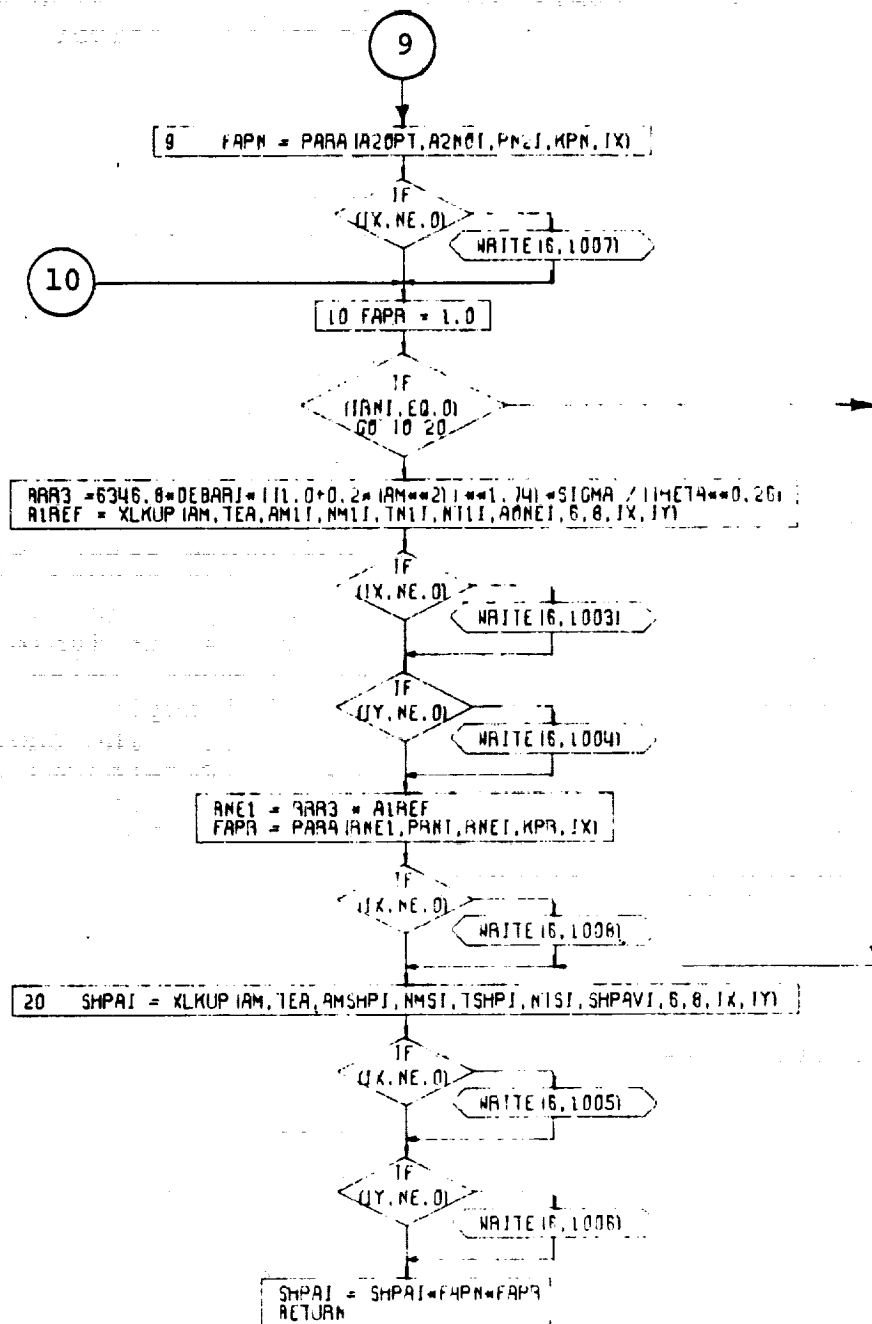


Figure 4-12. ENG 1 I Subroutine Flow Chart (Part 2 of 2).

4.5 ROTOR PERFORMANCE SUBROUTINE

Three options are available to the user for calculating rotor performance. These are specified by the indicator ROTIND as follows:

ROTIND

- | | |
|---|---|
| 1 | Rotor performance calculated by the "short form aero" rotor performance methodology |
| 2 | Rotor map is input, corrections are applied |
| 3 | Rotor map is input, no corrections are applied |

The first option (ROTIND = 1), using the short form aero methodology, allows the user to calculate rotor performance for a wide range of rotors with a minimum amount of input. The user is required to input a rotor cycle (a list of currently available cycles is illustrated in Table 2-4) and such blade characteristics as blade number, twist, and cut out. In the case of a single rotor helicopter, tail rotor blade characteristics must also be input. The short form aero methodology, developed at Boeing (References 2, 3, and 4), combines momentum theory and empirical corrections through coefficients found in the rotor cycles. The data used in this approach has been derived and correlated for rotors operating within the following parametric ranges:

Blade Number	=	2 - 6
Blade Twist	=	0 - -15°
Blade Root Cutout	=	0.20R
Rotor Solidity	=	0.055 - 0.150
Rotor Advance Ratio (μ)	=	0 - 0.4

No appreciable loss in accuracy is likely for cases involving more than six blades, less than 20 percent root cut out or a solidity lower than 0.055. The level of confidence will be reduced, however, for those cases in which the rotor parameters greatly exceed the ranges shown above. Figure 4-13 illustrates a typical comparison of short form aero predicted performance and flight test data.

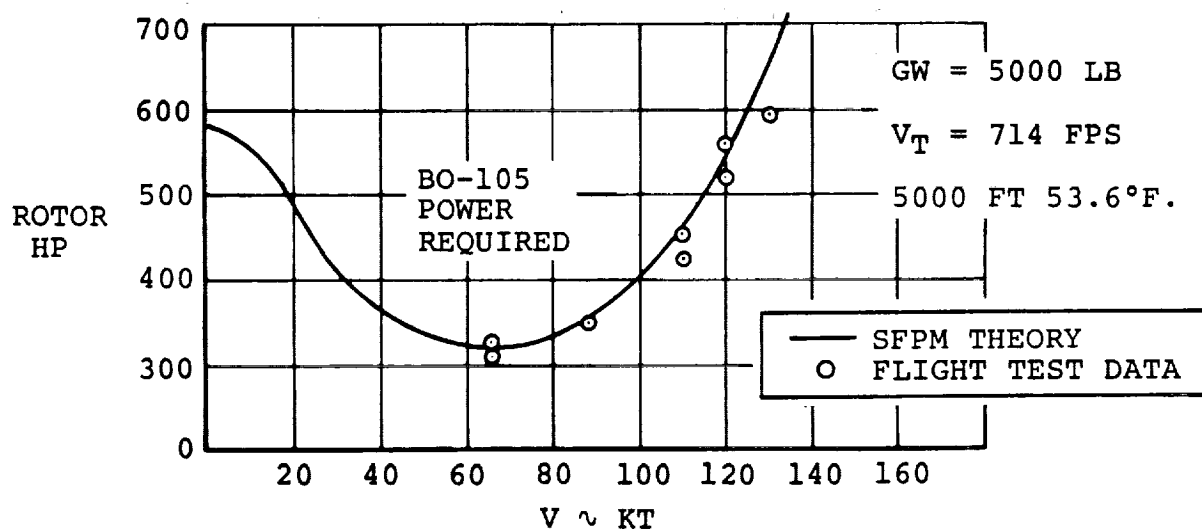
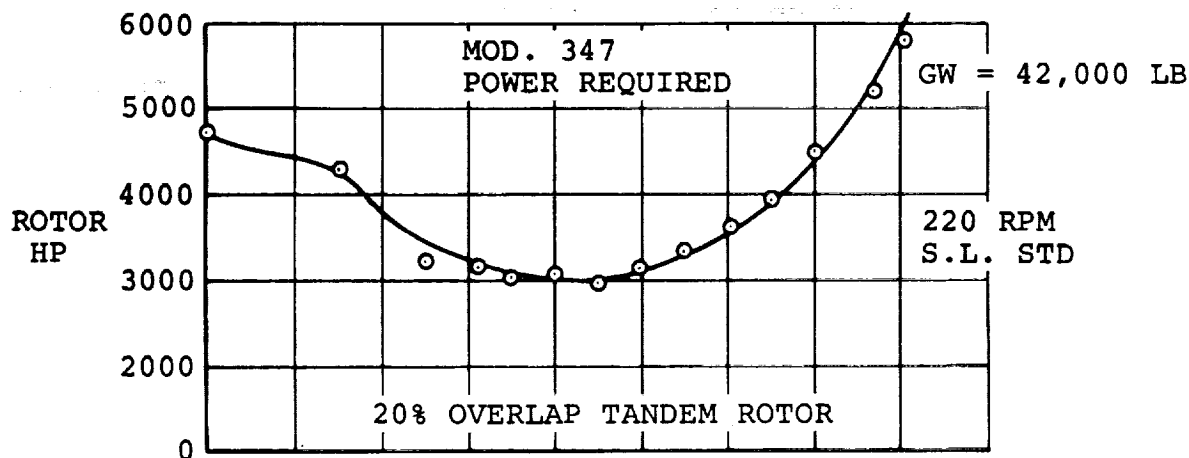
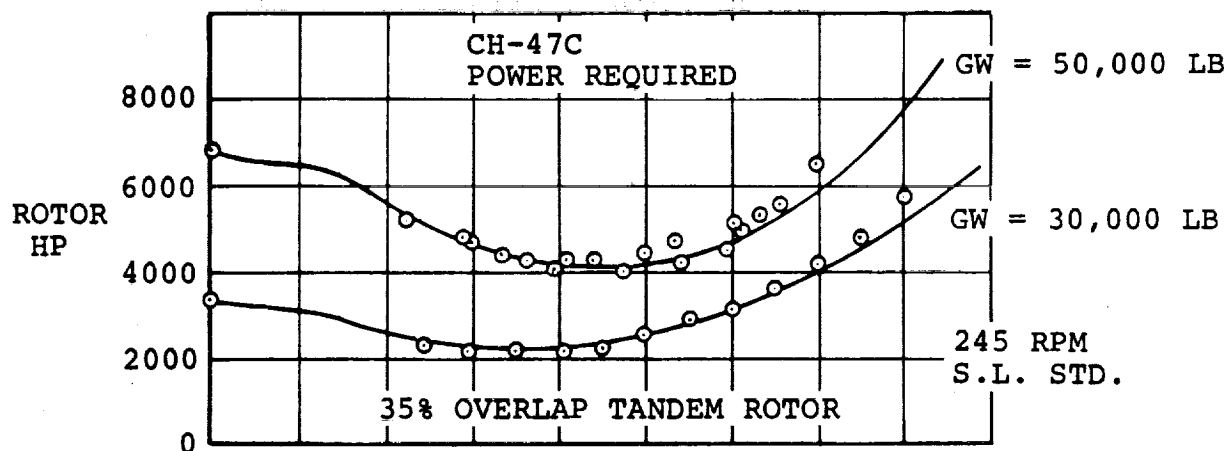


Figure 4-13. COMPARISON OF "SHORT FORM AERO" ROTOR PERFORMANCE AND FLIGHT TEST DATA

The second option (ROTIND = 2) utilizes isolated rotor data derived for a specified rotor configuration, but corrected by the program for the specific rotor and helicopter configuration being analyzed. It should be noted that this option, in the case of the single rotor helicopter, utilizes the short form aero methodology for calculating tail rotor power. Thus, the same tail rotor blade information required in the first option must be input.

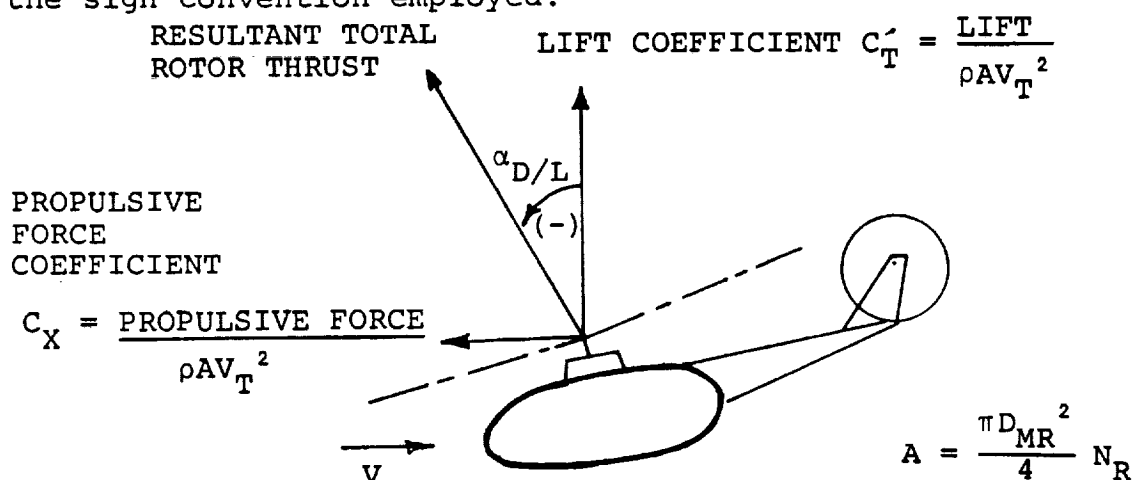
Option three (ROTIND = 3) uses total configuration rotor data (i.e. in the case of a single rotor helicopter, this would include both main and tail rotor power) and applies no corrections to the data. Input locations 2700-3410 are provided for the input of these rotor maps. Values of C_p/σ input as functions of up to ten values of C_T/σ at up to six values of M_{TIP} can be used for hover performance; and cruise C_p/σ values can be input as functions of up to ten values of C_T'/σ and ten values of C_X/σ at up to six values of μ .

For the calculation of vertical climb power, the subroutine uses the simple potential energy relationship:

$$RHP_{VRC} = \frac{W (V_{RC})}{33000 [V_{CEH_1} + V_{CEH_2} (V_{RC})]}$$

The vertical climb efficiency factors (V_{CEH_1} and V_{CEH_2}) can be derived from flight test data.

The quantity ALPHA D/L printed out in all forward flight performance segments reflects the propulsive thrust-lift vector of the main rotor. The simple sketch below illustrates the sign convention employed.



The flow chart for this subroutine is illustrated by Figure 4-14.



Figure 4-14. ROTPOW Subroutine Flow Chart (Part 1 of 14).

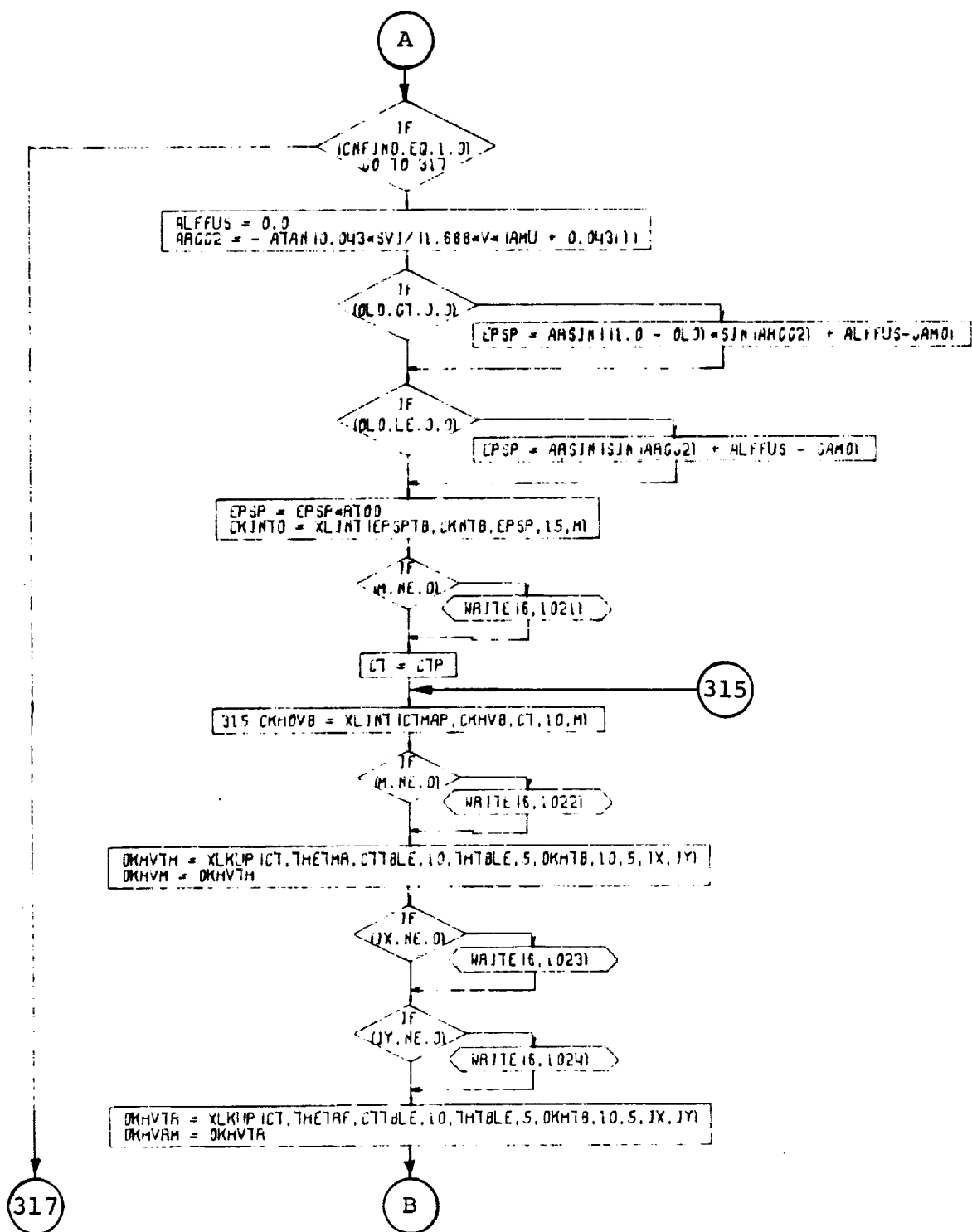


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 2 of 14).

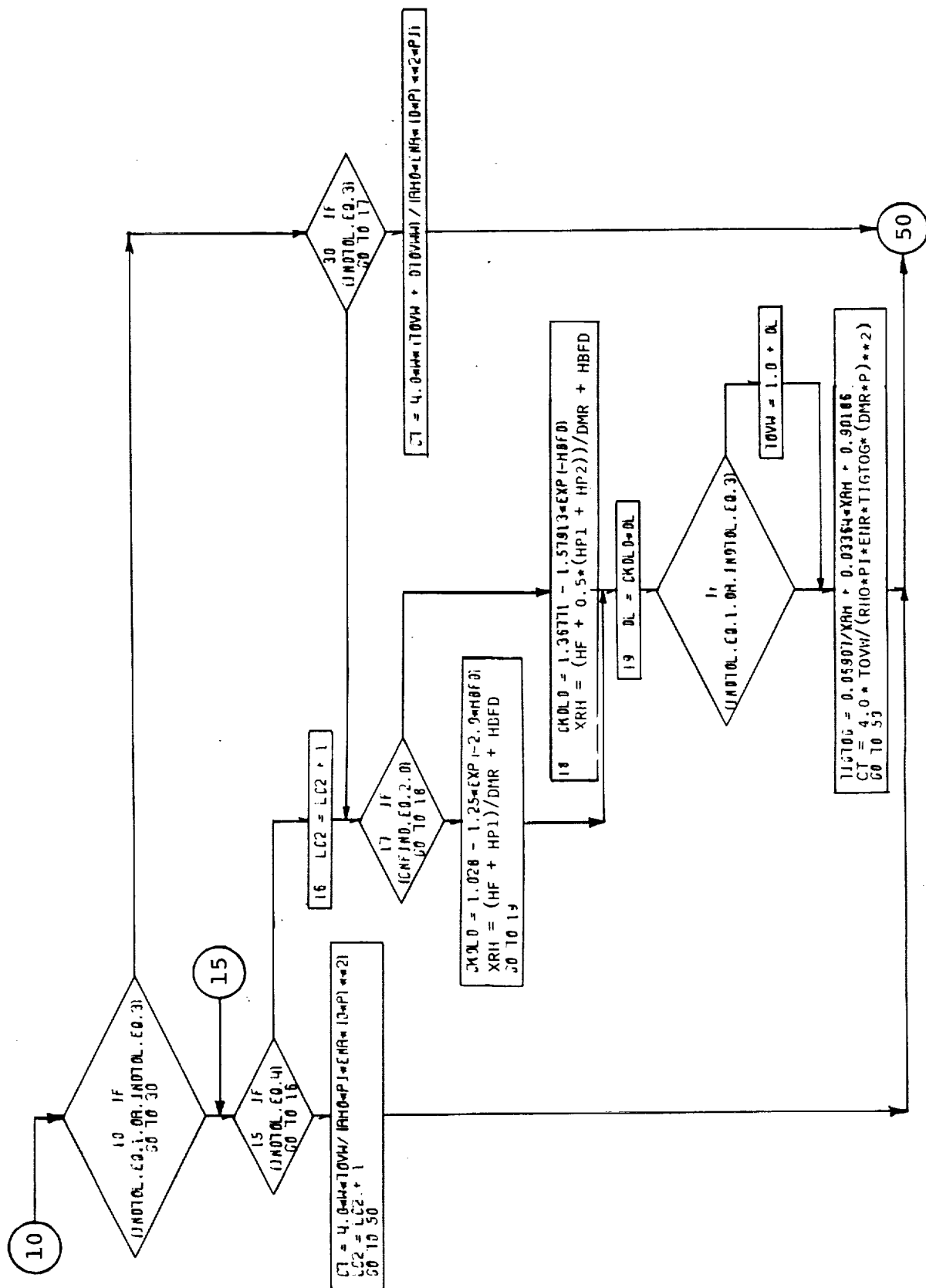


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 4 of 14).

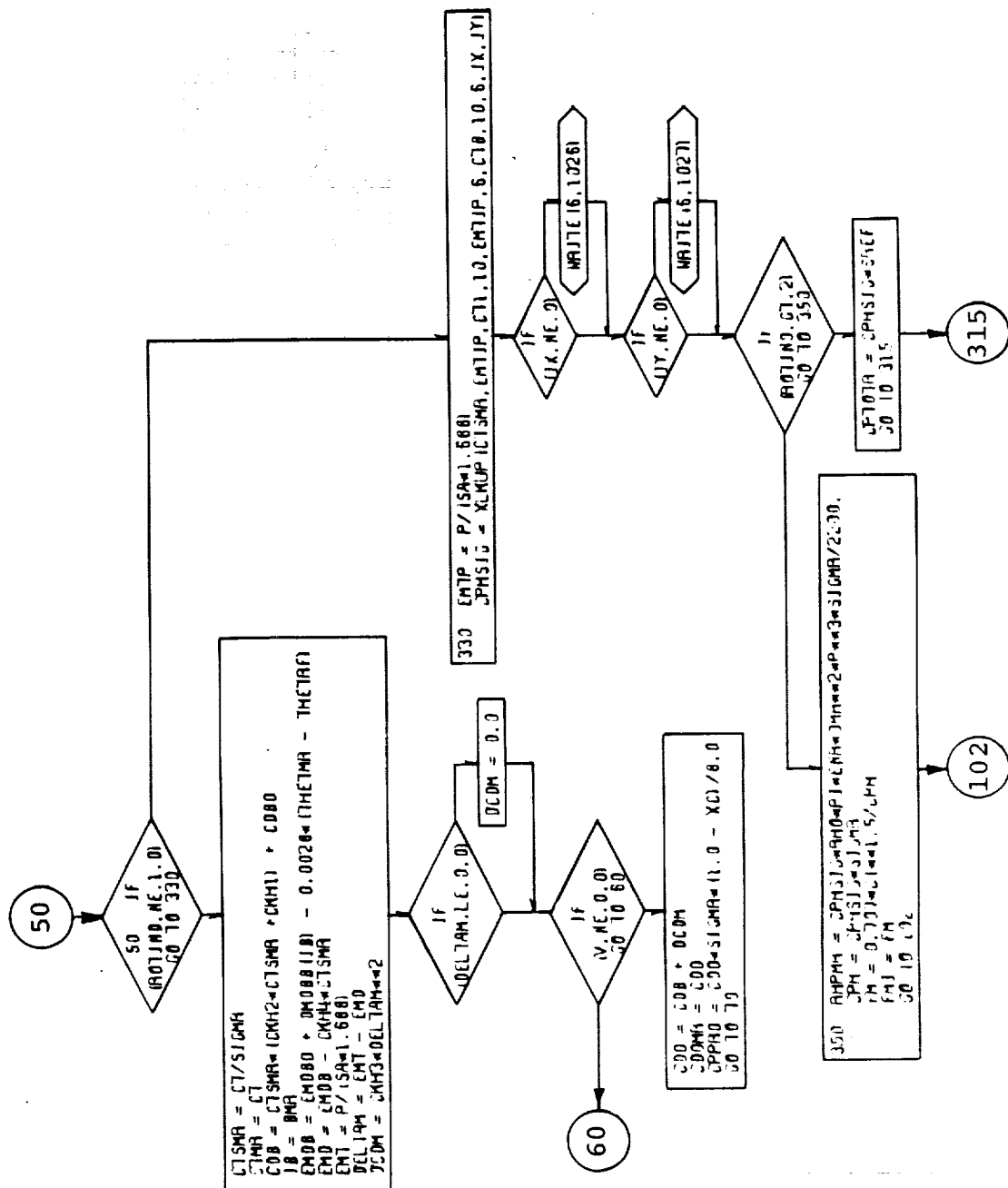


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 5 of 14).

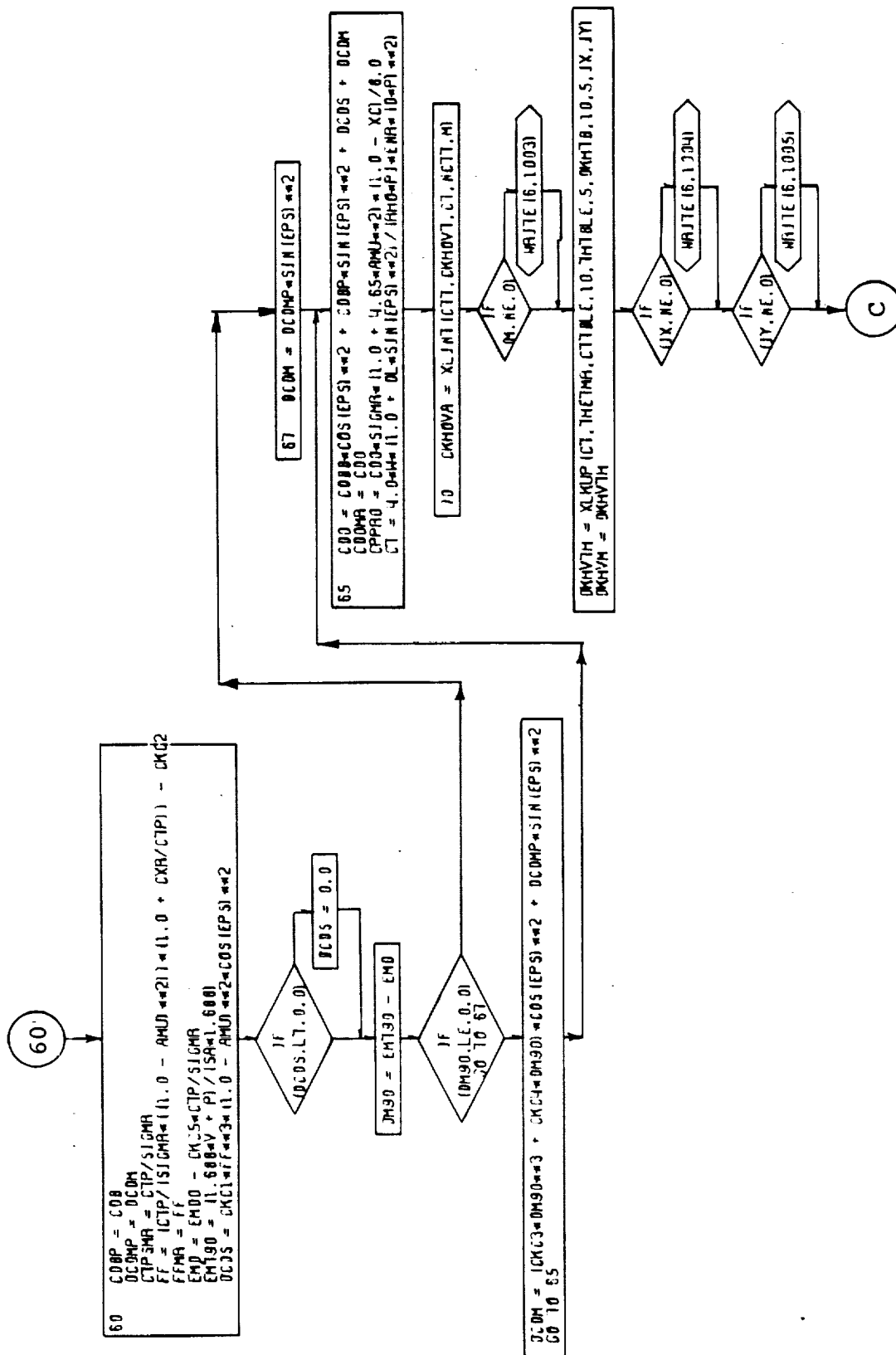


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 6 of 14) .

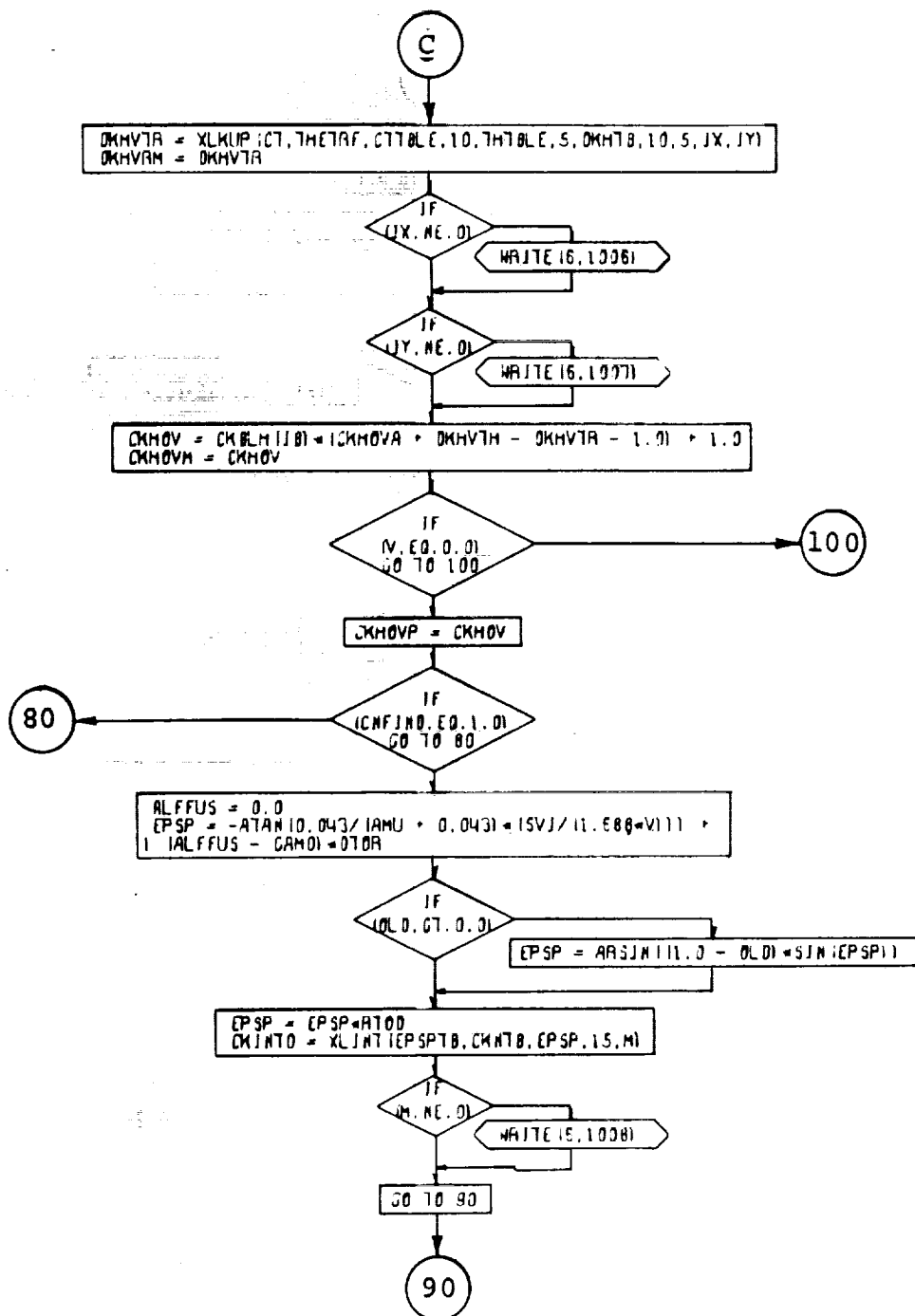


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 7 of 14).

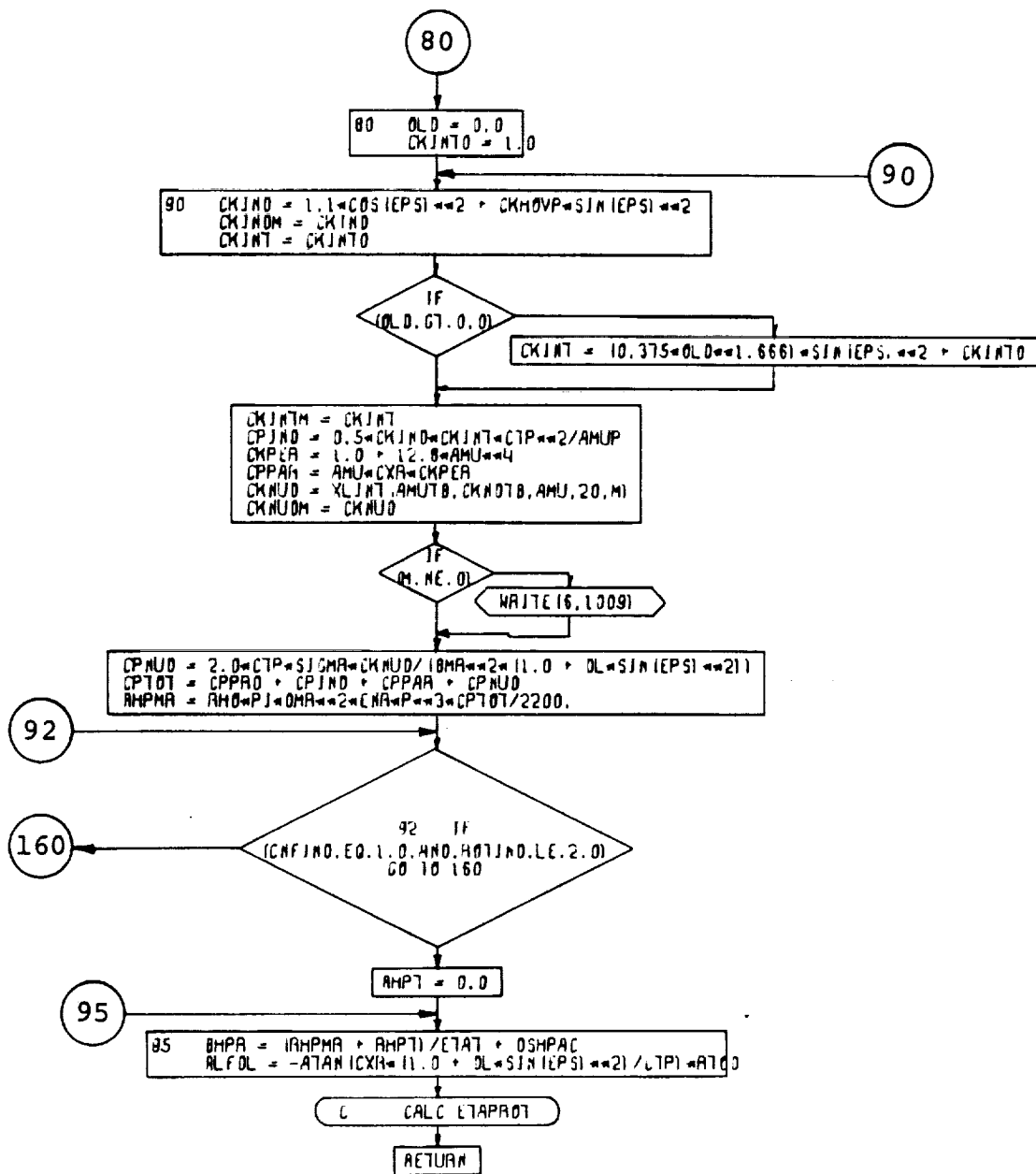


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 8 of 14).

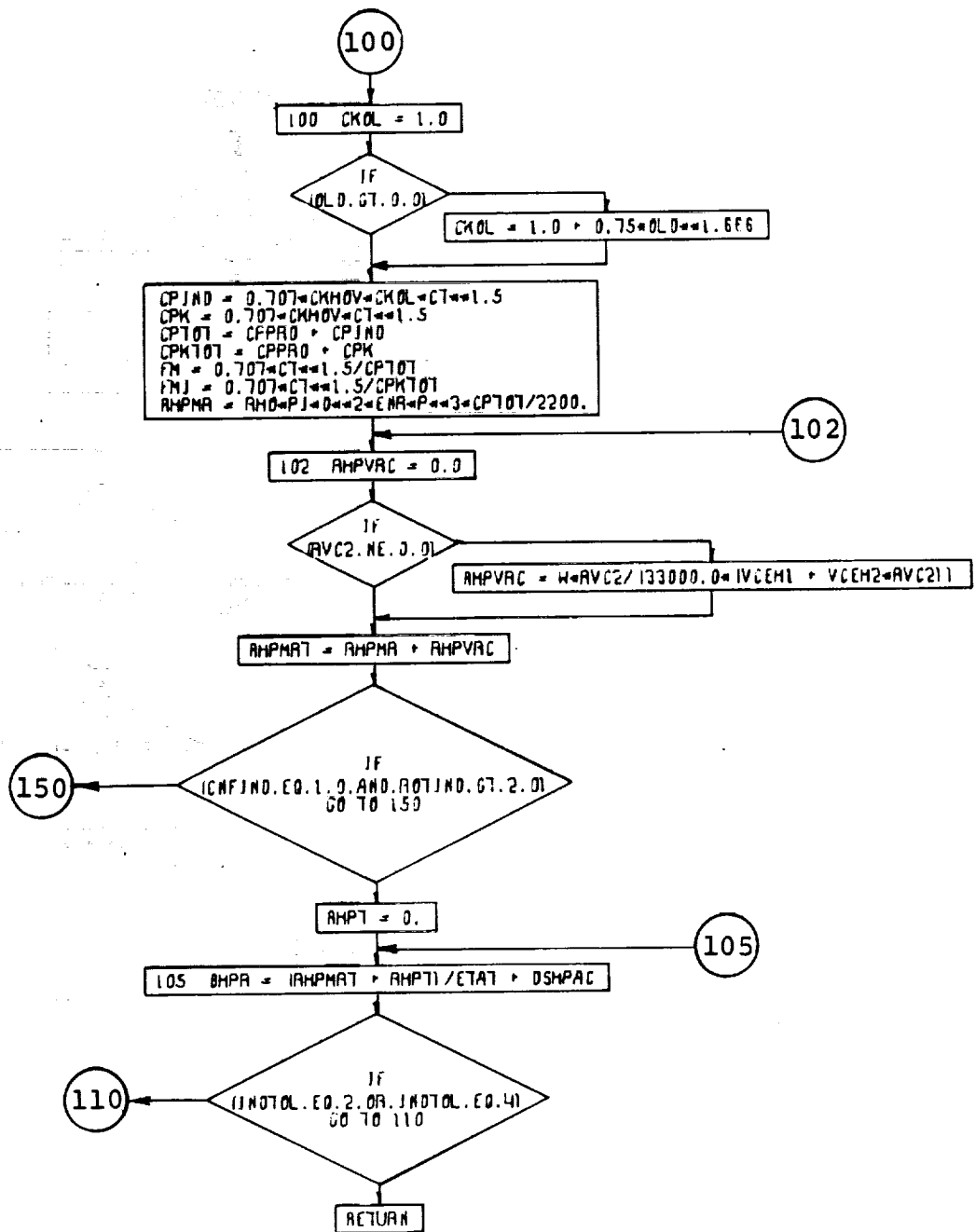


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 9 of 14).

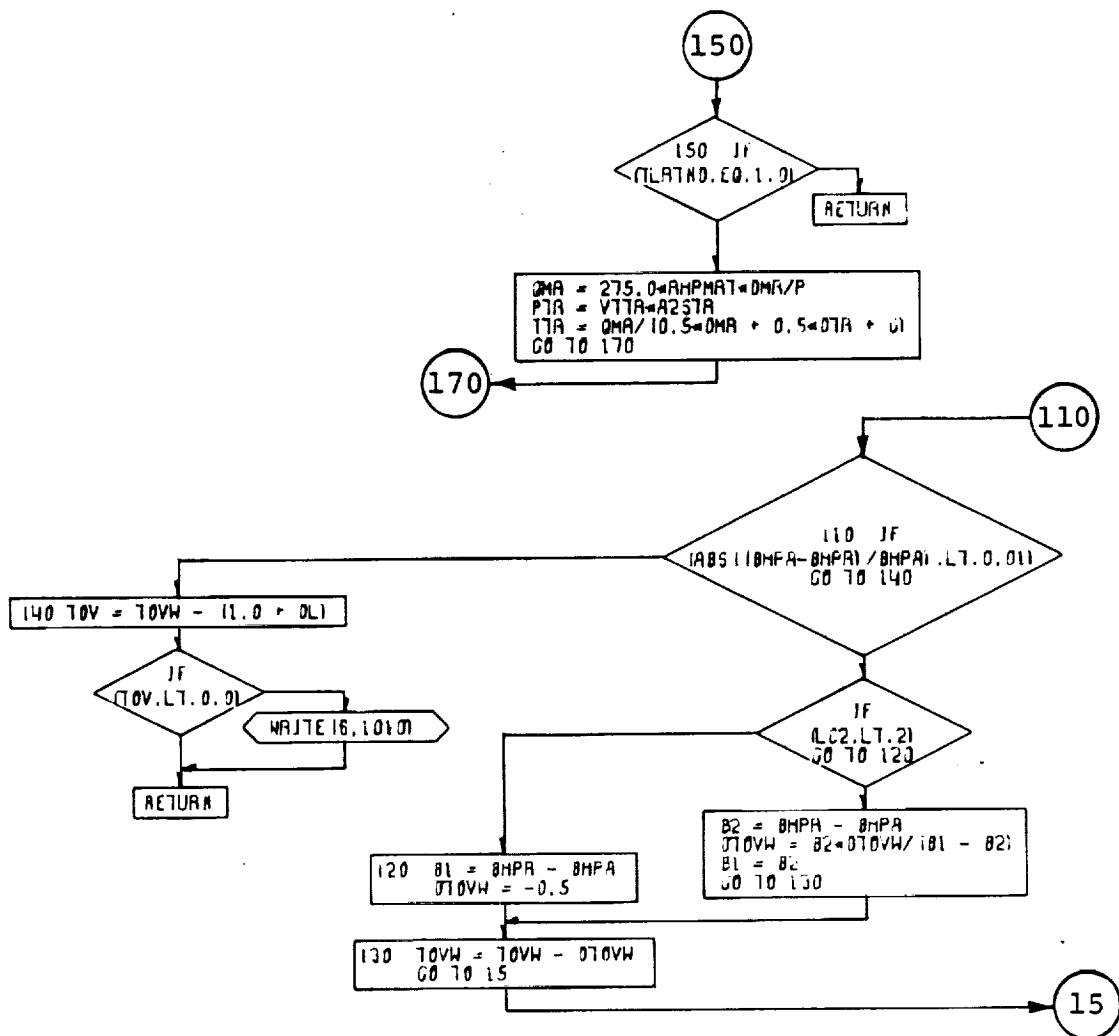


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 10 of 14).

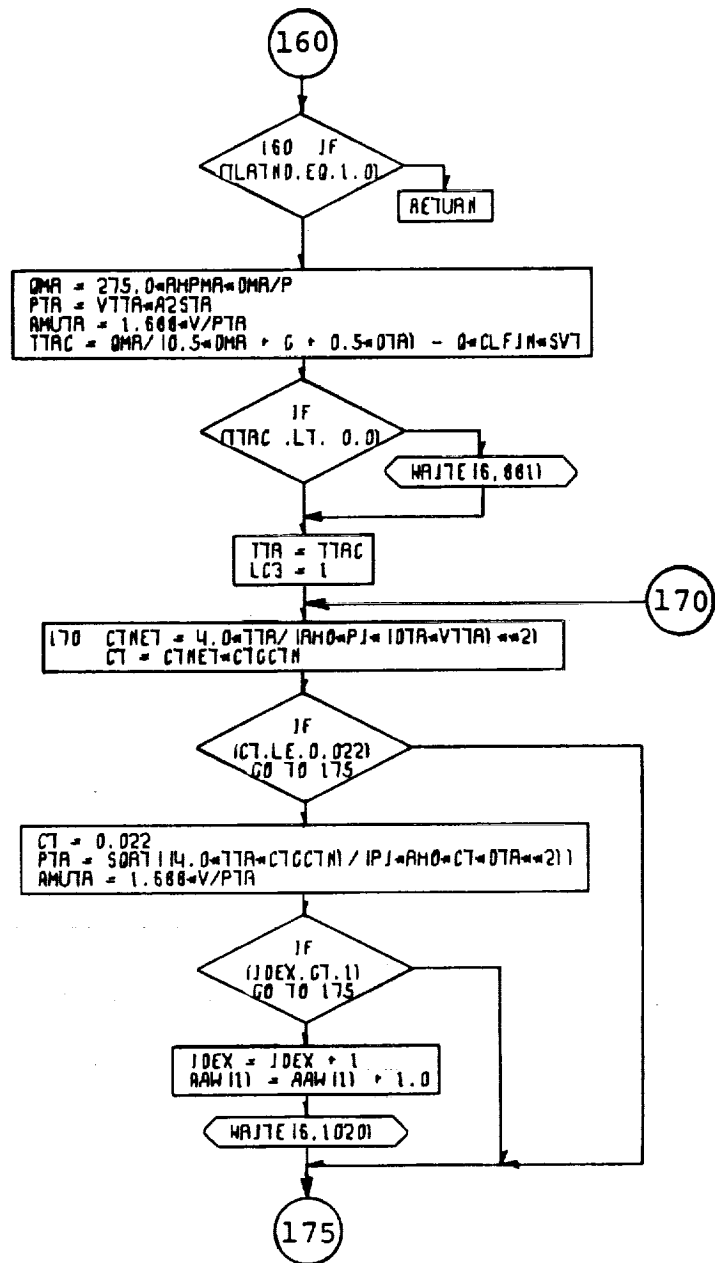


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 11 of 14).

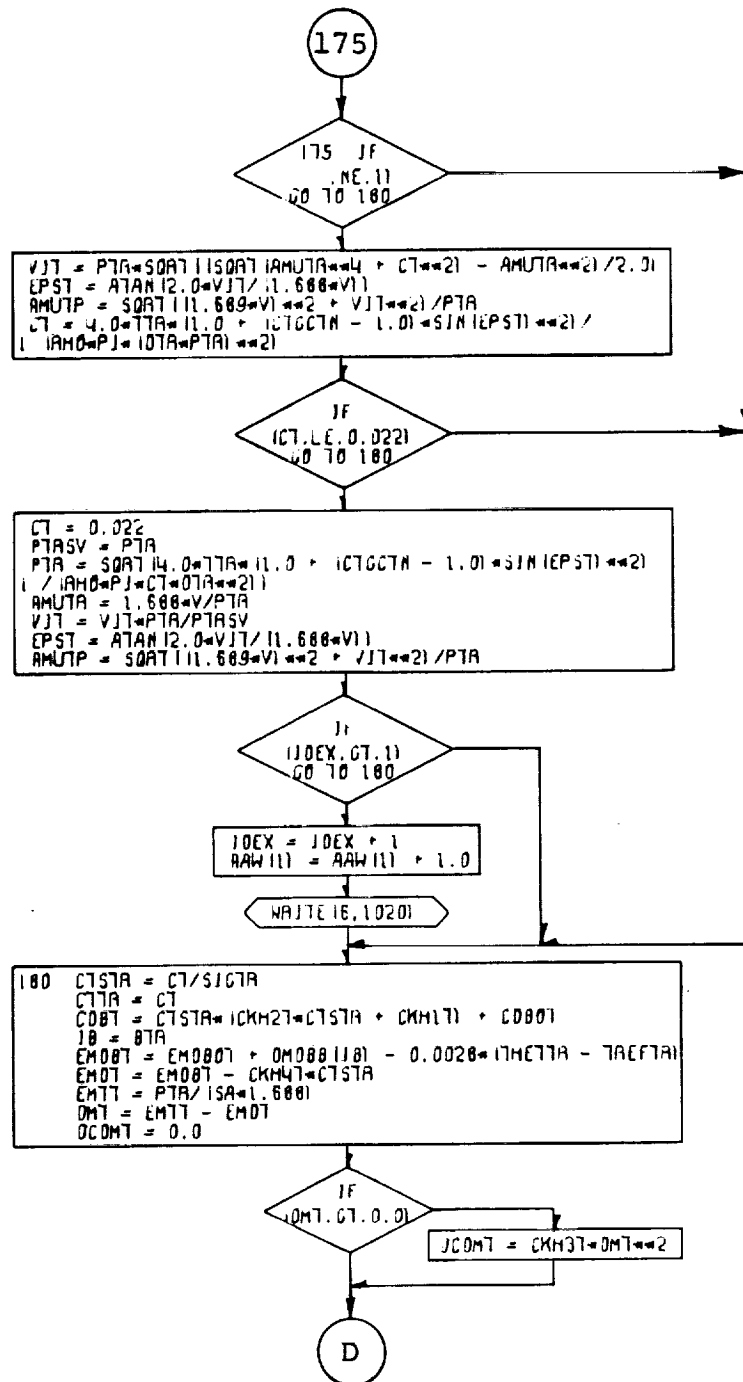


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 12 of 14).

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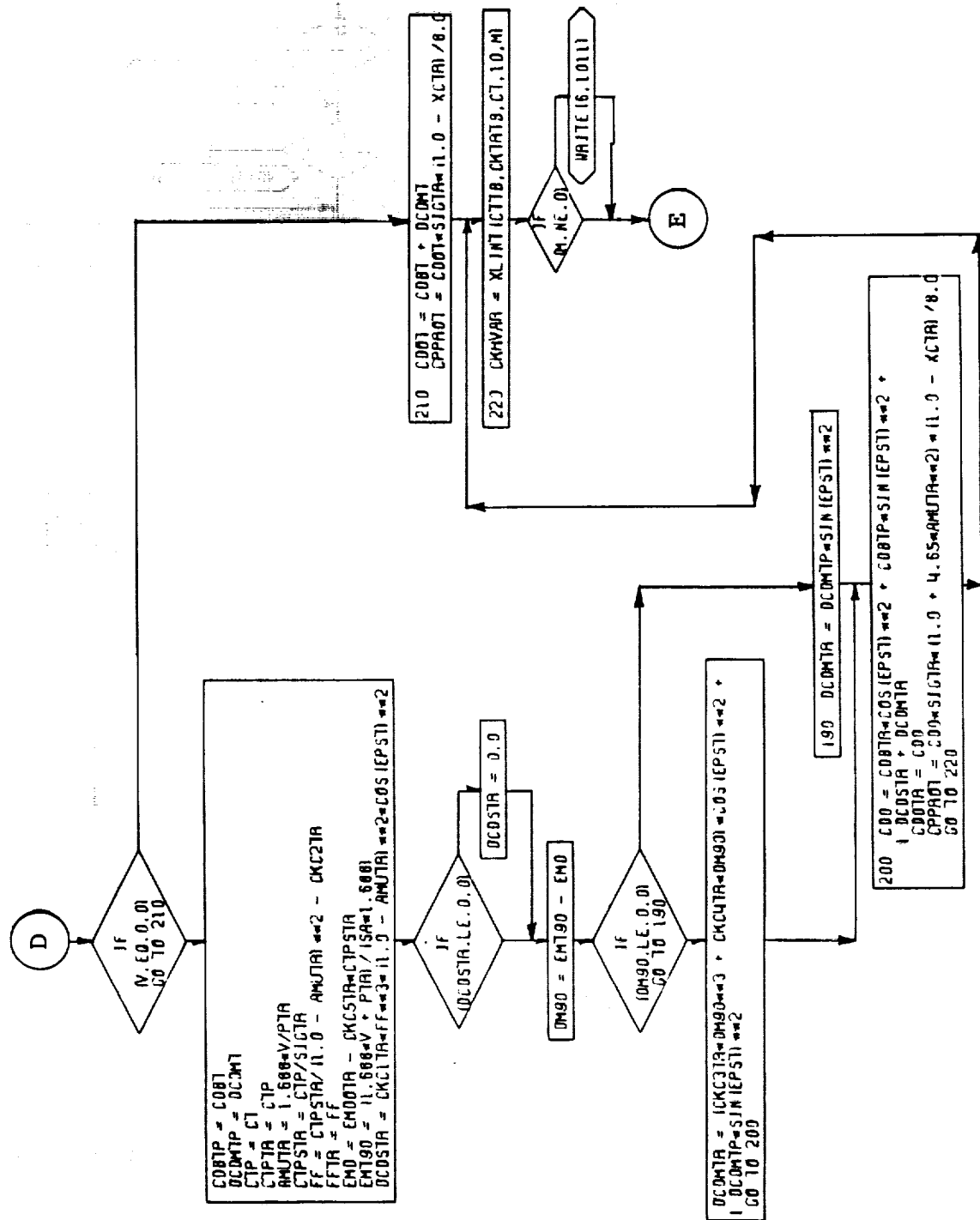


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 13 of 14).

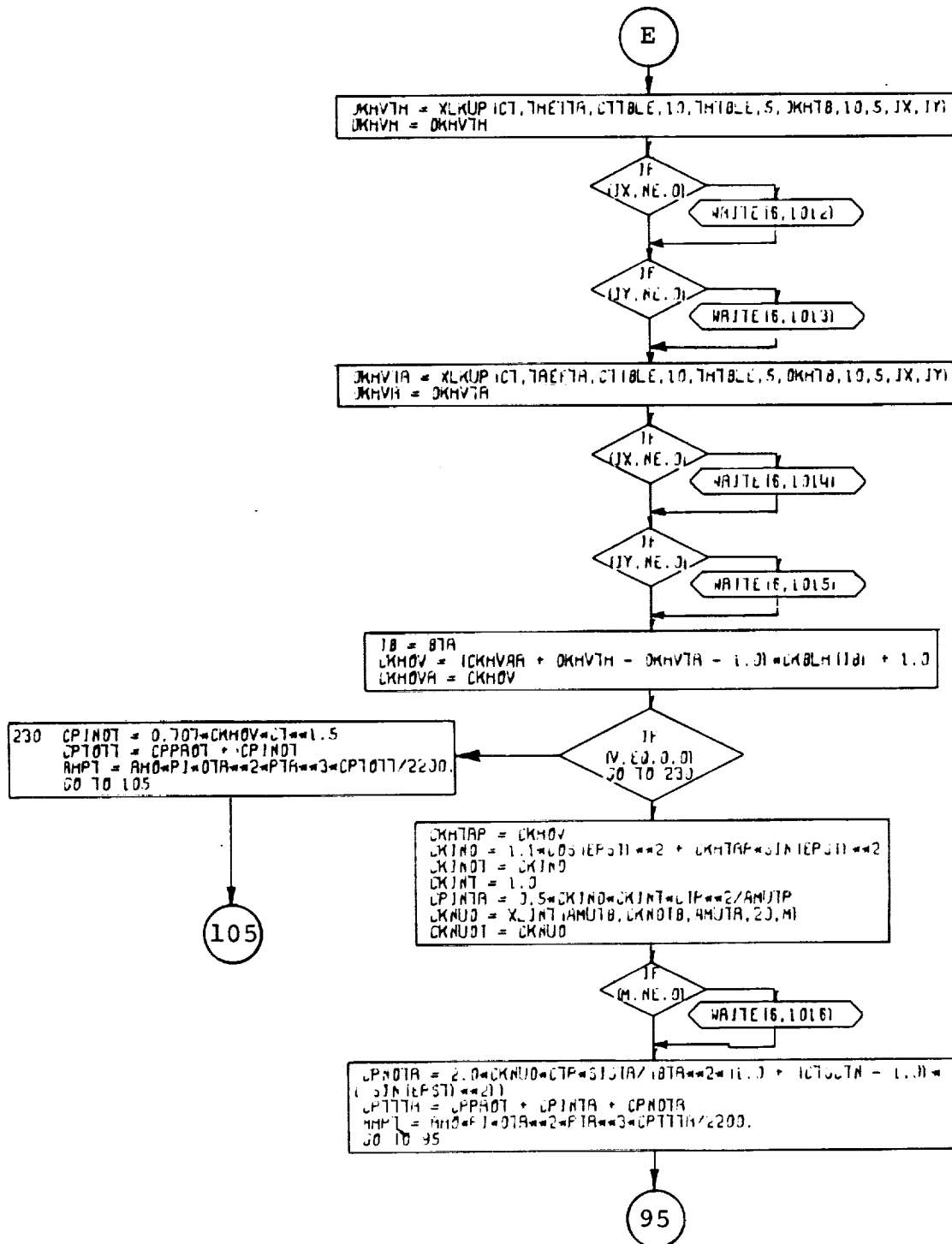


Figure 4-14. ROTPOW Subroutine Flow Chart (Part 14 of 14).

4.6 ROTOR LIMITS SUBROUTINE

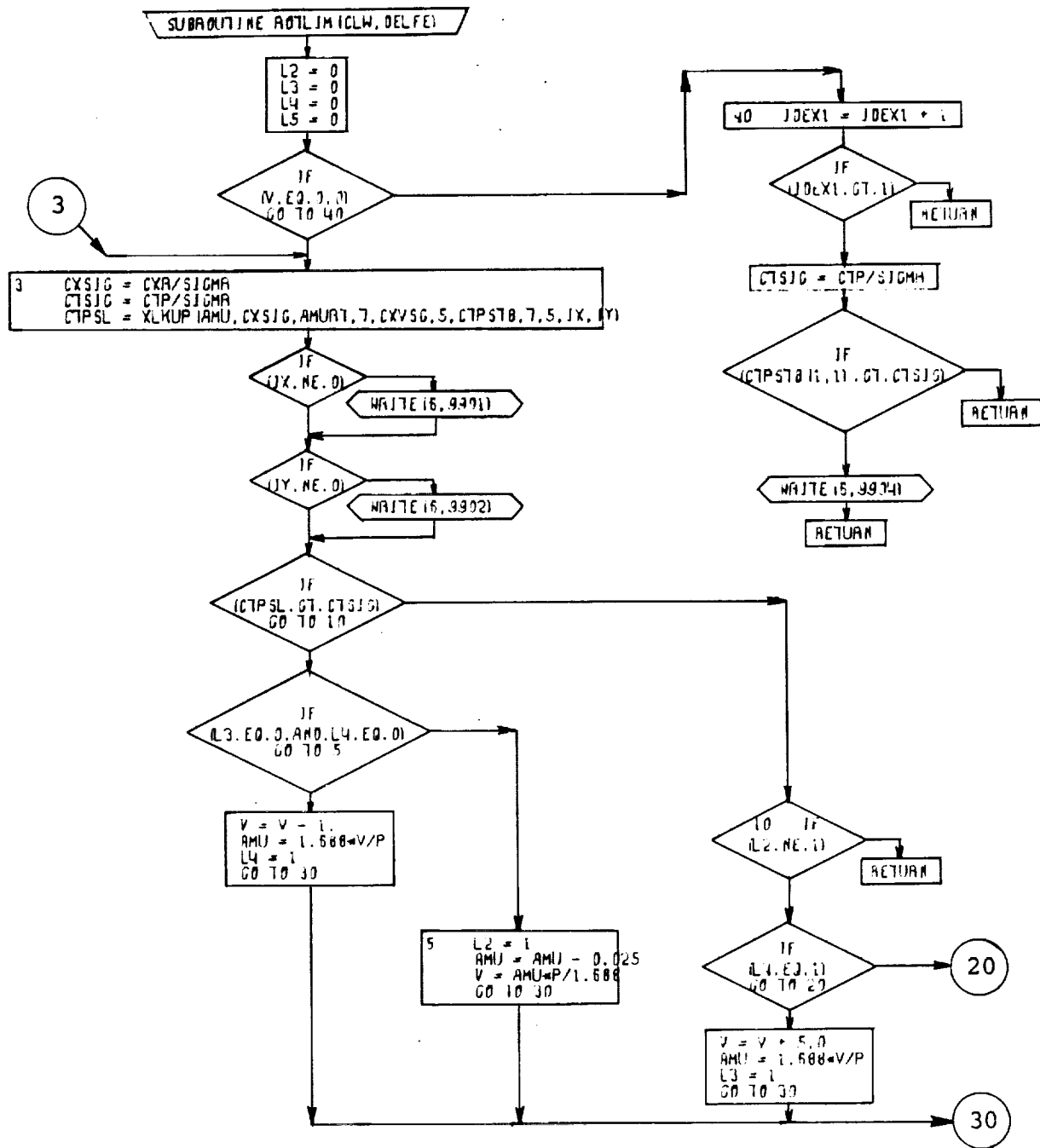
The rotor limits subroutine compares the main rotor operating values of μ , $\frac{C_{XR}}{\sigma}$, and C_T'/σ to those input in the rotor limits information table (LOC 0347-0395). In the takeoff, hover, and landing subroutine, if the main rotor operating value of C_T'/σ exceeds the table value, the following statement is printed out:

WARNING: ROTOR LIMIT HAS BEEN EXCEEDED. EITHER
REDUCE MAIN ROTOR THRUST REQUIREMENTS AT
THESE OPERATING CONDITIONS, OR INCREASE
MAIN ROTOR TIP SPEED. CHECK ALL VALUES
OF C_T'/σ IN THIS PERFORMANCE LEG.

In the climb, cruise, descent, and loiter subroutines, if the main rotor operating value of C_T'/σ for a given $\frac{C_{XR}}{\sigma}$ and μ exceeds the table value, cruise speed is reduced until the operating and table values of C_T'/σ coincide and the following message is printed out:

WARNING: ROTOR LIMIT HAS BEEN EXCEEDED. FORWARD
FLIGHT SPEED HAS BEEN REDUCED ACCORDINGLY.
CHECK ALL VALUES OF TAS, MU, C_T'/σ , AND
CXR IN THIS PERFORMANCE LEG.

Section 7.3 provides a more detailed discussion of Rotor Limits. Figure 4-15 is a flow chart of this subroutine.



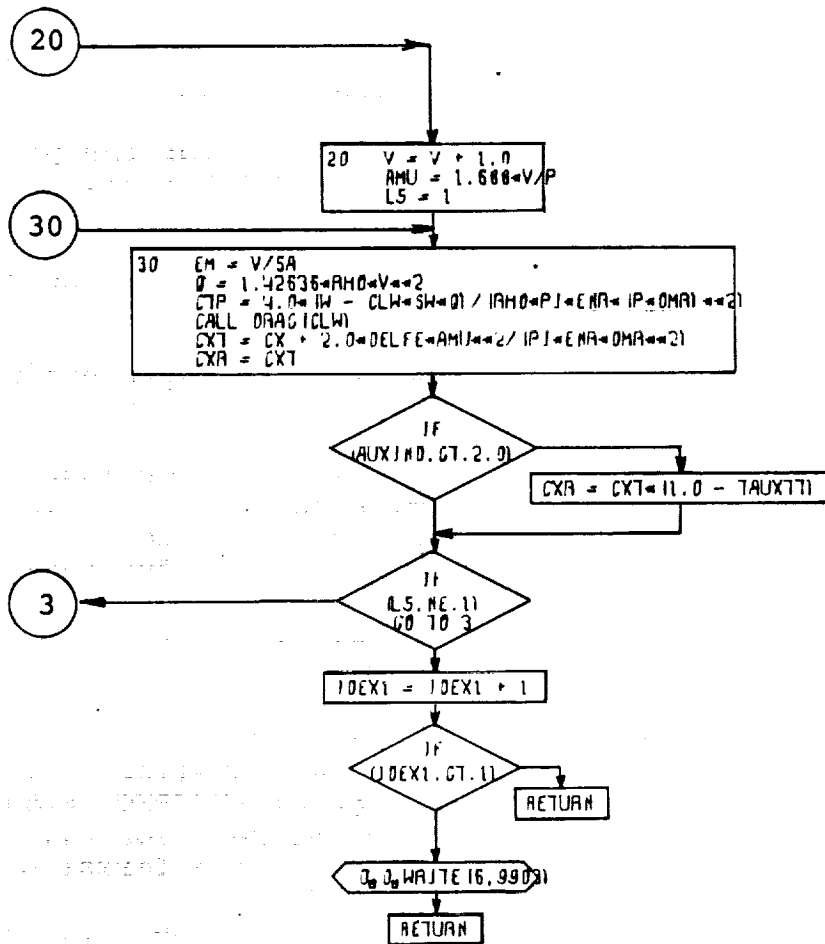


Figure 4-15. ROTLIM Subroutine Flow Chart (Part 2 of 2).

4.7 PROPELLER PERFORMANCE CALCULATIONS

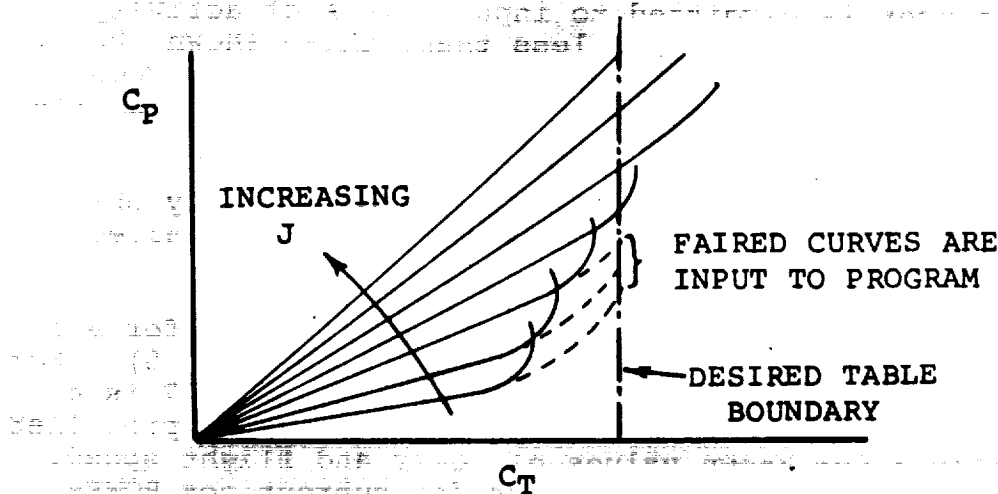
Three different options are available for representing the performance of propellers when using turboshaft engines (ENGIND = 0). The option to be used is specified to the program by means of a prop efficiency indicator - " η_p IND".

η_p IND = 0 - The user inputs a set of point values for the prop efficiency for the performance segments of climb and descent and a table of efficiency as a function of flight Mach number for cruise and loiter. The following input is required:

- η_{p3} - A single point value is input for the prop efficiency during climb (SGTIND = 3).
- η_{p4} - A table is input of prop efficiency during cruise (SGTIND = 4) and Loiter (SGTIND = 6) as a function of flight Mach number.
- η_{p5} - A single point value is input representing the prop efficiency during Descent (SGTIND = 5).

The primary advantage of this option of propeller performance representation is that it permits rapid evaluation of the sensitivity of aircraft performance and size to changes in propeller performance. It may also prove desirable to use this option in early conceptual studies when a specific prop has not been picked and it is desired to use "reasonable" values of efficiency.

η_p IND = 1 - This option permits the user to input a table representing the performance of the propeller throughout the flight envelope with the exception of DESCENT (SGTIND = 5) for which a value of η_{p5} is input as before. For all other performance segments the table, input in the format of C_p (prop power coefficient) as a function of C_T (prop thrust coefficient) and J (advance ratio), is used. The table which is prepared must include all compressibility losses for the known tip speed at which the propeller is intended to operate. The user is cautioned that the tabular values must be monotonic. That is, the table cannot include the maximum in C_T which reflects blade stall at high values of C_p . This must be faired out as shown in the sketch at the top of the next page.



The advantage of this option is that it permits the user to input the performance of a real propeller as determined from test data.

$\eta_{pIND} = 2$ - Through use of this option the program will automatically calculate the performance of a wide variety of V/STOL propellers. The user need only specify the number of blades (3 or 4), the activity factor per blade, and the integrated lift coefficient, C_{L_i} . The method used for the calculation of propeller performance is the "short method" originated at the Curtiss-Wright Corporation's Propeller Division (Reference 10). The method involves the use of a set of equations which can be developed from strip theory. These equations permit the propeller performance maps (C_p , C_T , J) to be transformed into an "equivalent" lift-drag polar for the propeller. Conversely, the lift-drag polars, once developed, can be used with the equations to predict the propeller performance. For incompressible flow, the "equivalent" lift-drag polar which is used depends only on the value of C_{L_i} being considered. That is, for a given C_{L_i} the same polar can be used to accurately represent the performance of props with a wide variation in activity factor and number of blades and for a wide range of C_p and J . For compressible flow conditions, the curves correlate very well on the basis of the value of helical Mach number at the 3/4 radial station. The equivalent lift-drag polars which are contained in the program were developed from detailed strip analysis calculations for cruise. These detailed calculations covered the following range of parameters:

Number of blades:	3 and 4
Activity factor/blade:	60 → 220
Integrated lift coefficient, C_{L_i} :	0.15 → 0.7

Although the user is permitted to input values of activity factor and C_{Li} greater than (or less than) those shown above, the level of confidence in the predictions is reduced when values for those parameters are outside the range used in the detailed calculations.

Figure 4-16 is characteristic of the level of accuracy obtained from the short method when compared to the detailed calculations.

This option will calculate the propeller performance for all mission performance segments except Descent (SGTIND = 5). For Descent, the user inputs a value for η_{p5} . Figure 4-17 is a flow chart of subroutine THRUST which calculates the propeller thrust available for known values of power and flight speed. Figures 4-18 and 4-19 are flow charts for subroutines POWER and POWERI in which the power required for specified thrust and flight speed is calculated. These subroutines make use of propeller equivalent lift-drag polars, as mentioned above, to calculate the performance of the propeller. The polars are developed in the main control loop for the particular value of integrated lift coefficient, C_{Li} , being studied from the following equations:

$$\gamma = \tan^{-1} (C_D/C_L) = \text{function of } M_H, C_L, C_{Li}$$

$$M_H = \text{helical Mach number @ } 3/4 \text{ } r/R$$

$$C_L = \text{equivalent lift coefficient at which prop is operating}$$

$$C_{Li} = \text{integrated lift coefficient of prop}$$

For cruise

$$\gamma = a_0 + a_1 C_{Li} + a_2 C_{Li}^2$$

a_0 , a_1 , and a_2 are coefficients stored in the program and are functions of M_H and C_L

The coefficients a_0 , a_1 , a_2 , are listed in Table 4-3.

The calculations of propeller performance for $\eta_{pIND} = 1$ and 2 are based on the assumption that the engines are interconnected by a cross shaft. That is, if engines are shut down during cruise and loiter the remaining power is evenly distributed to all of the propellers.

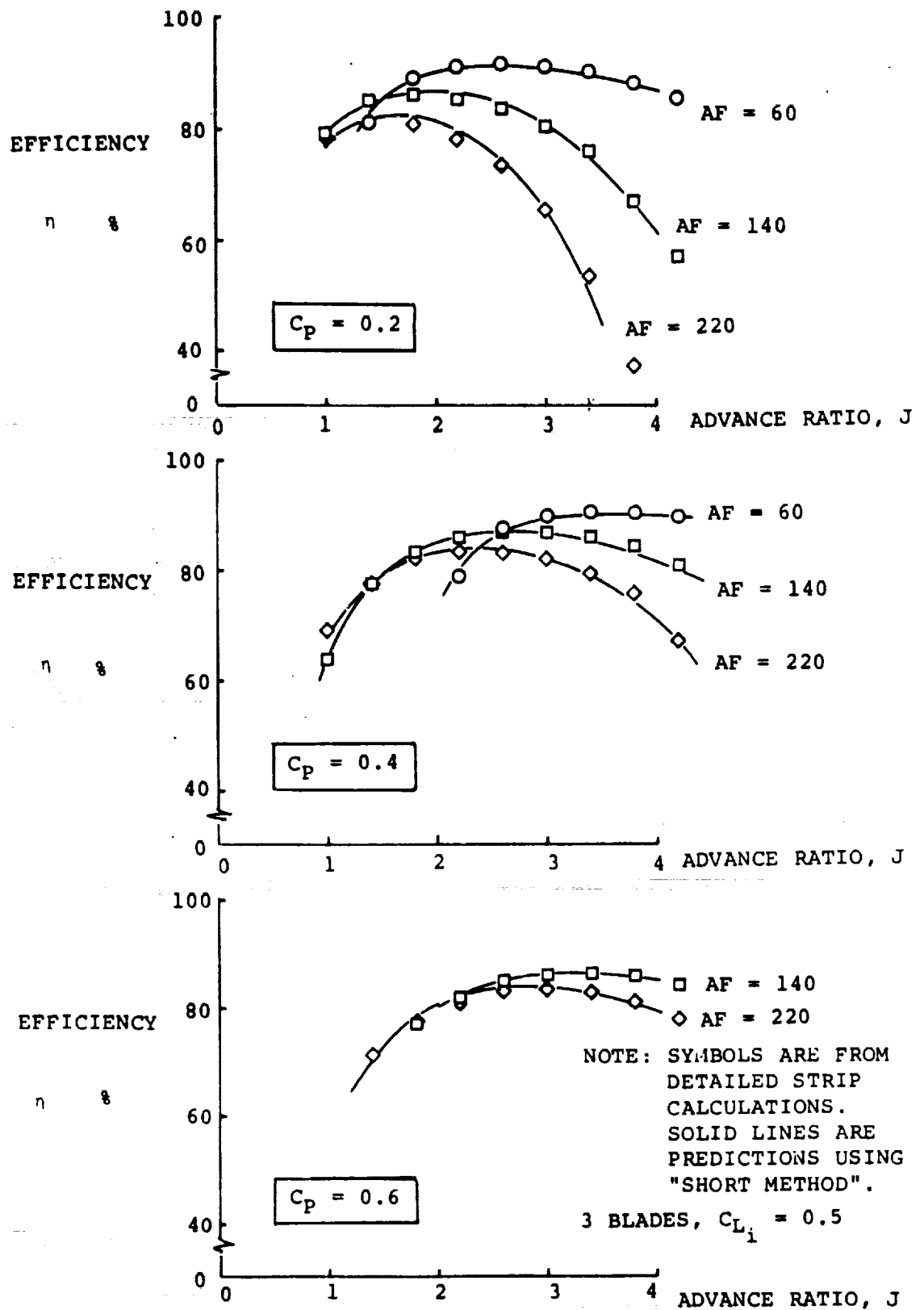


Figure 4-16. Comparison of "Short Method" and Detailed Calculations for Propeller Cruise Efficiency.

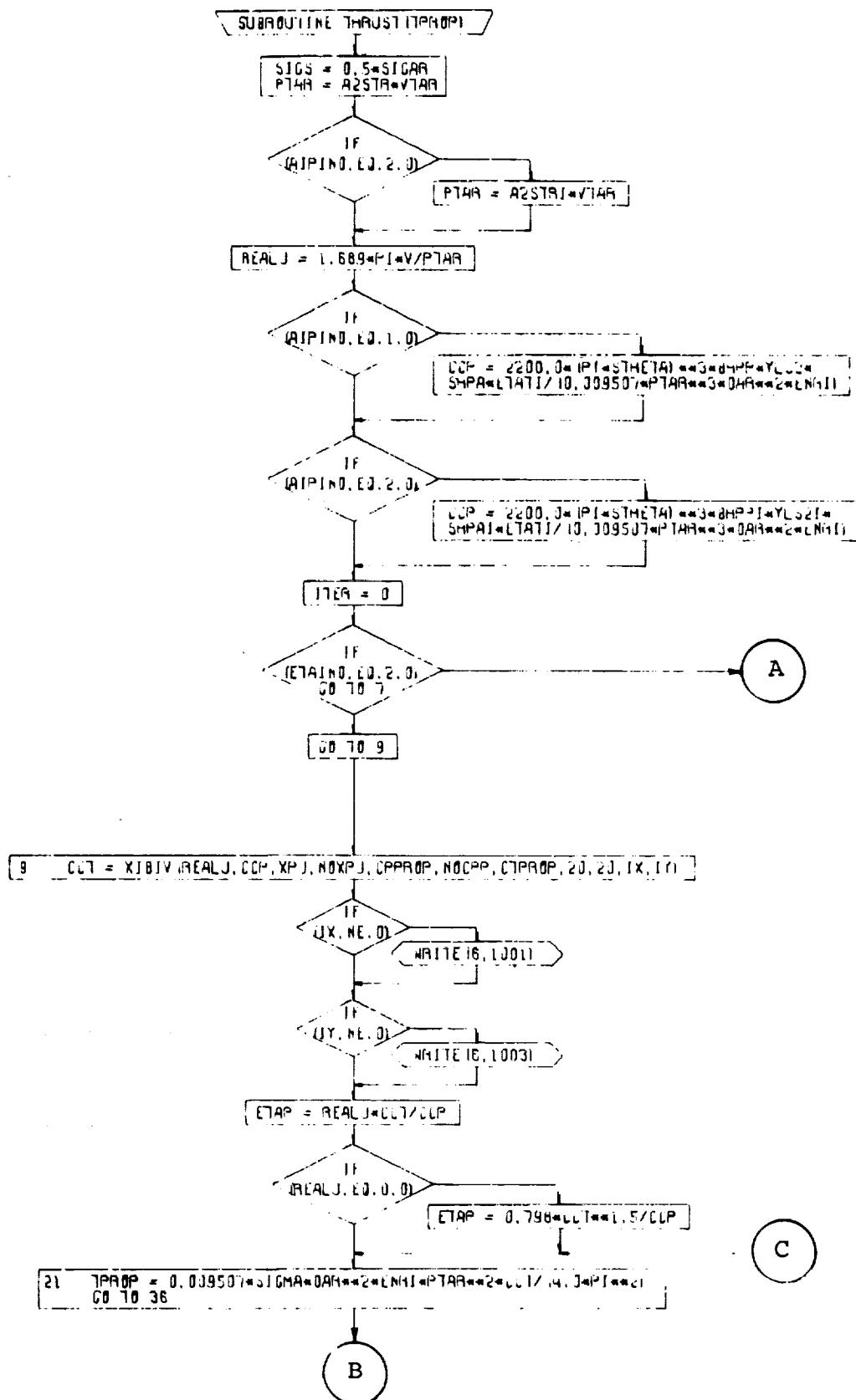


Figure 4-17. THRUST Subroutine Flow Chart (Part 1 of 4).

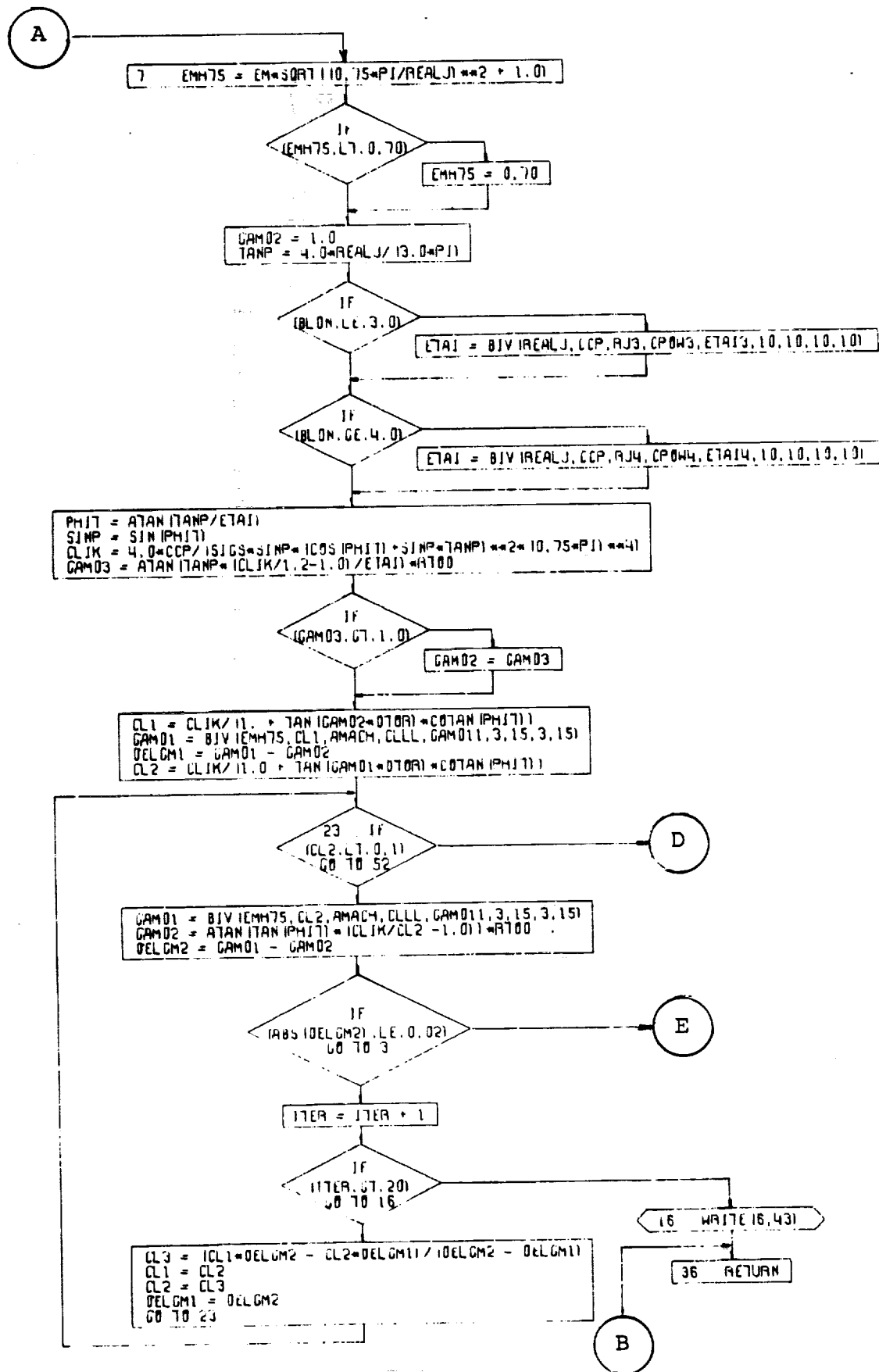


Figure 4-17. THRUST Subroutine Flow Chart (Part 2 of 4).



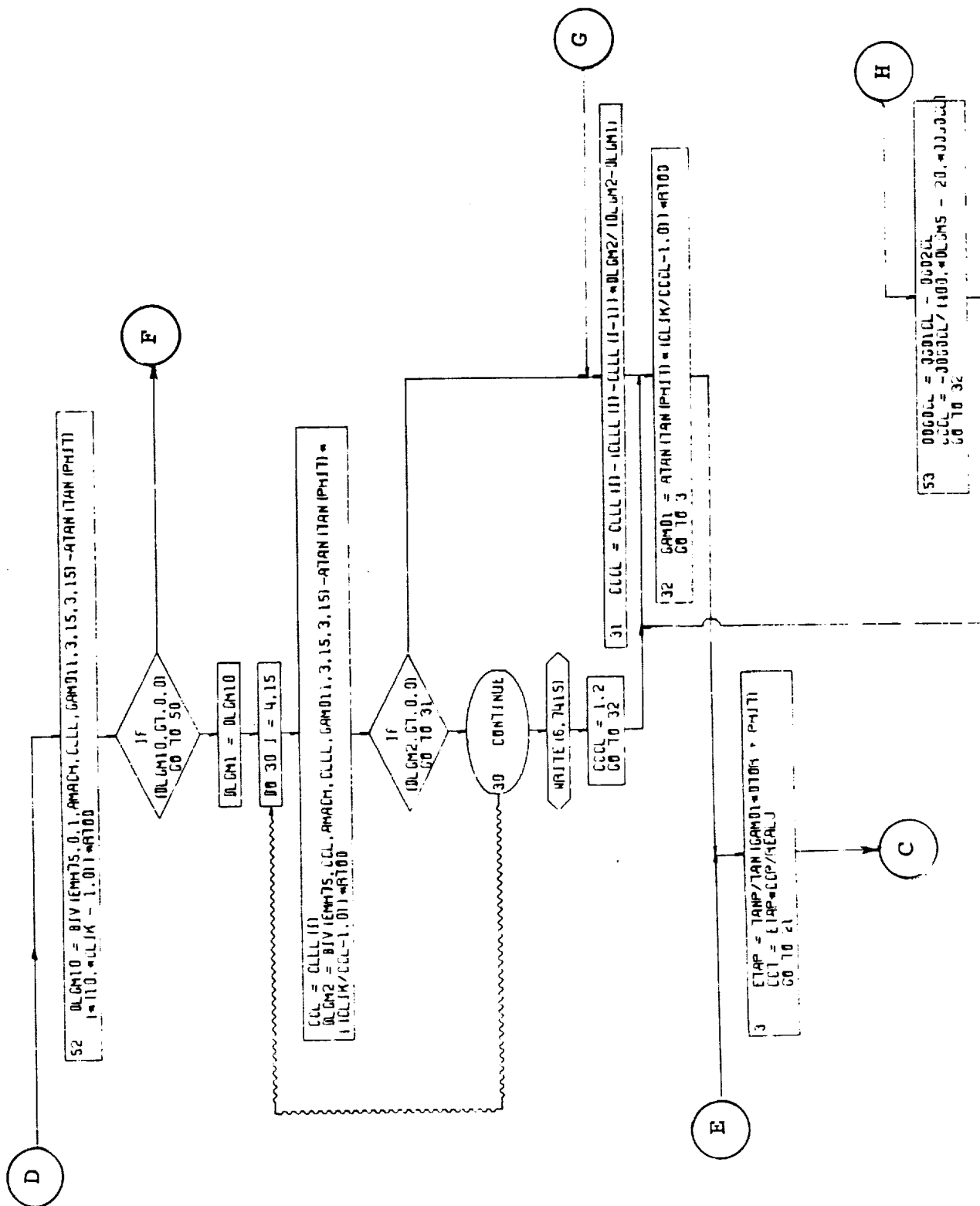


Figure 4-17. THRUST Subroutine Flow Chart (Part 3 of 4).

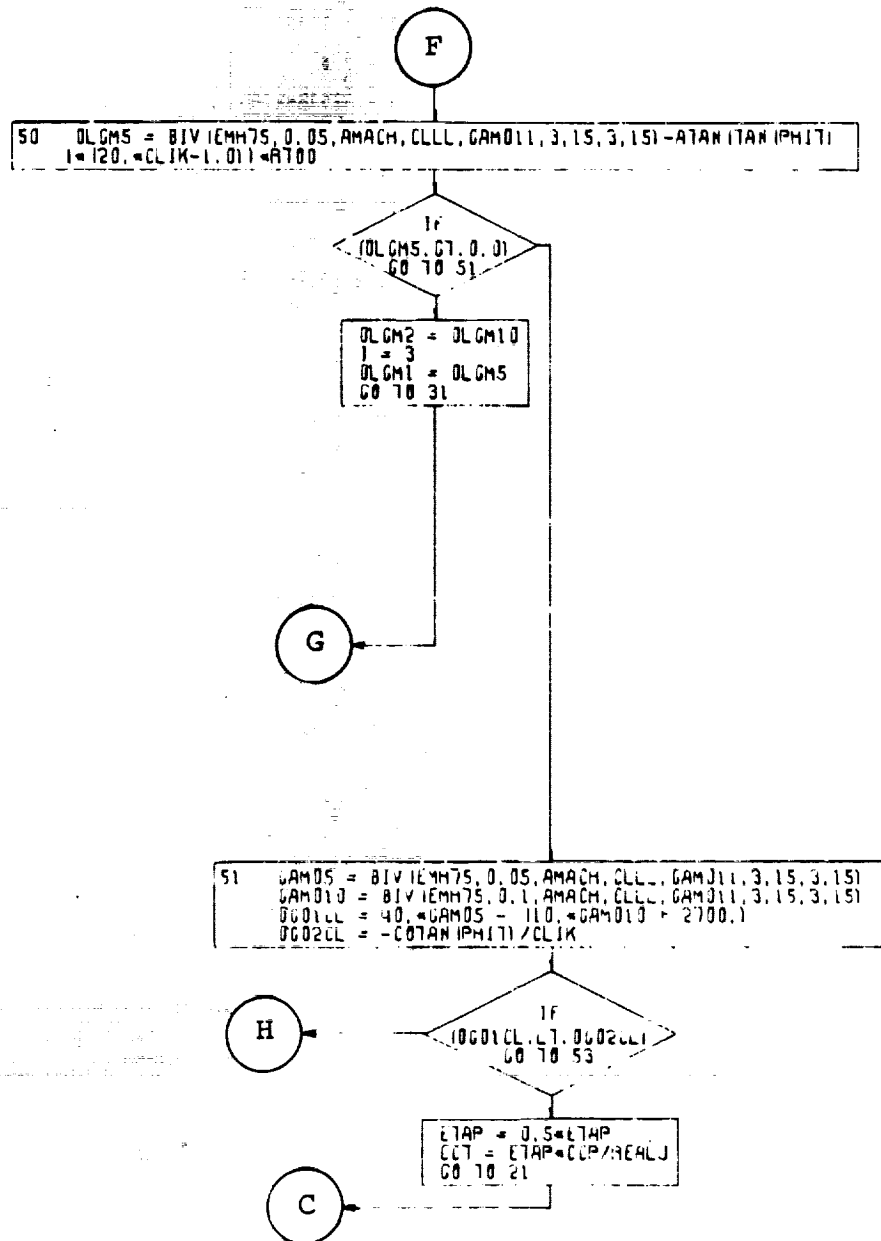


Figure 4-17. THRUST Subroutine Flow Chart (Part 4 of 4).

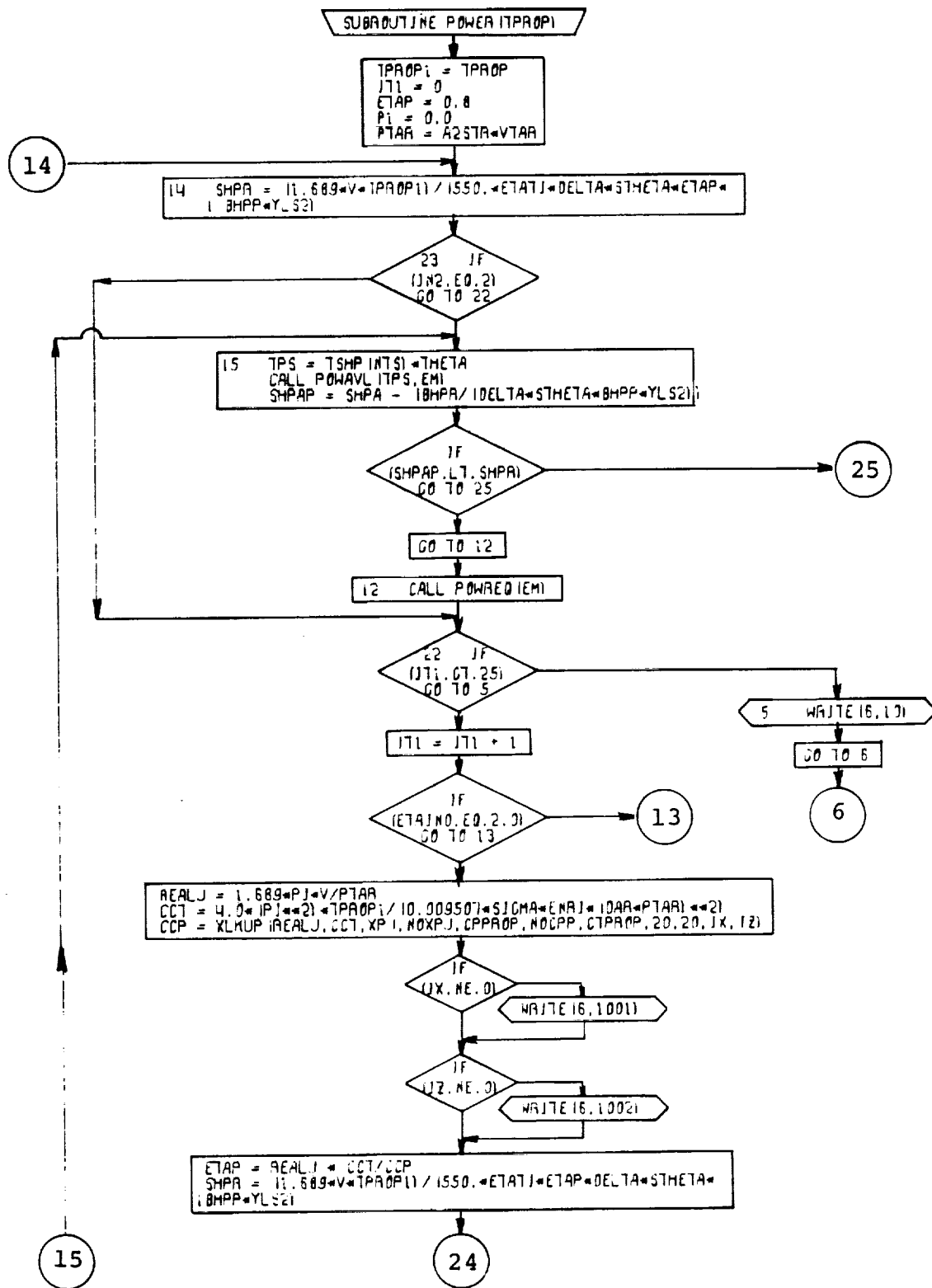


Figure 4-18. POWER Subroutine Flow Chart (Part 1 of 3).

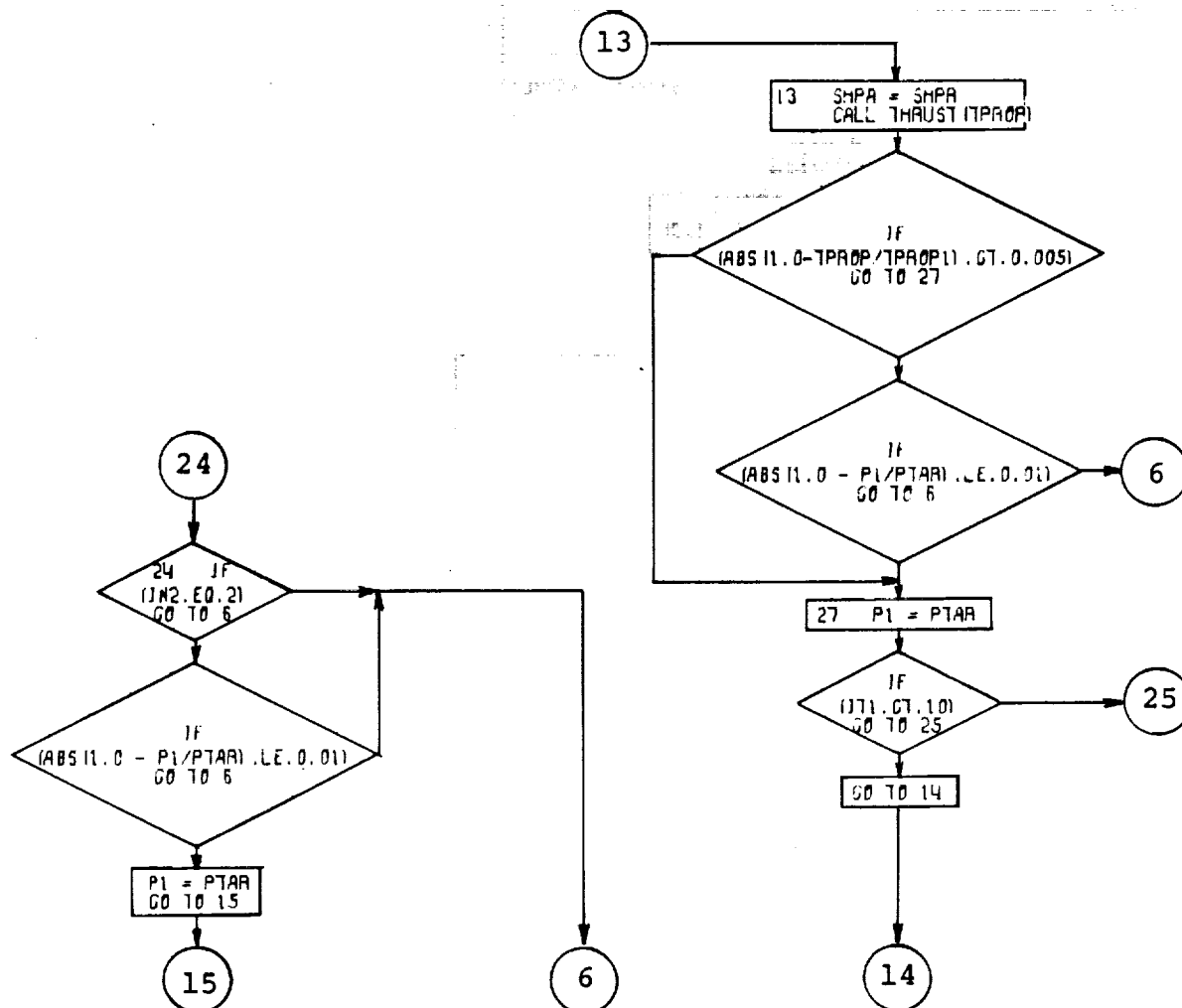


Figure 4-18. POWER Subroutine Flow Chart (Part 2 of 3).

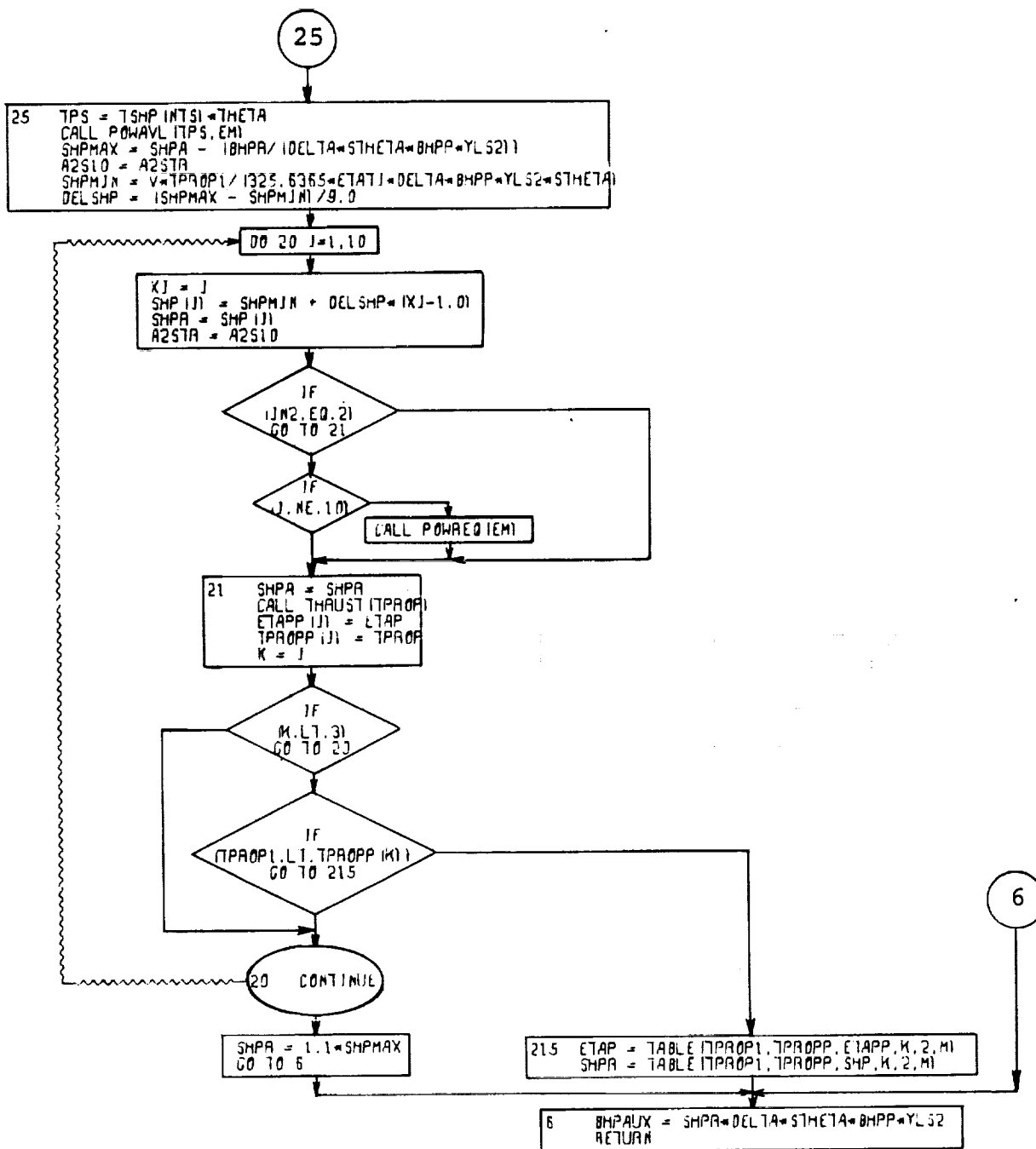


Figure 4-18. POWER Subroutine Flow Chart (Part 3 of 3).

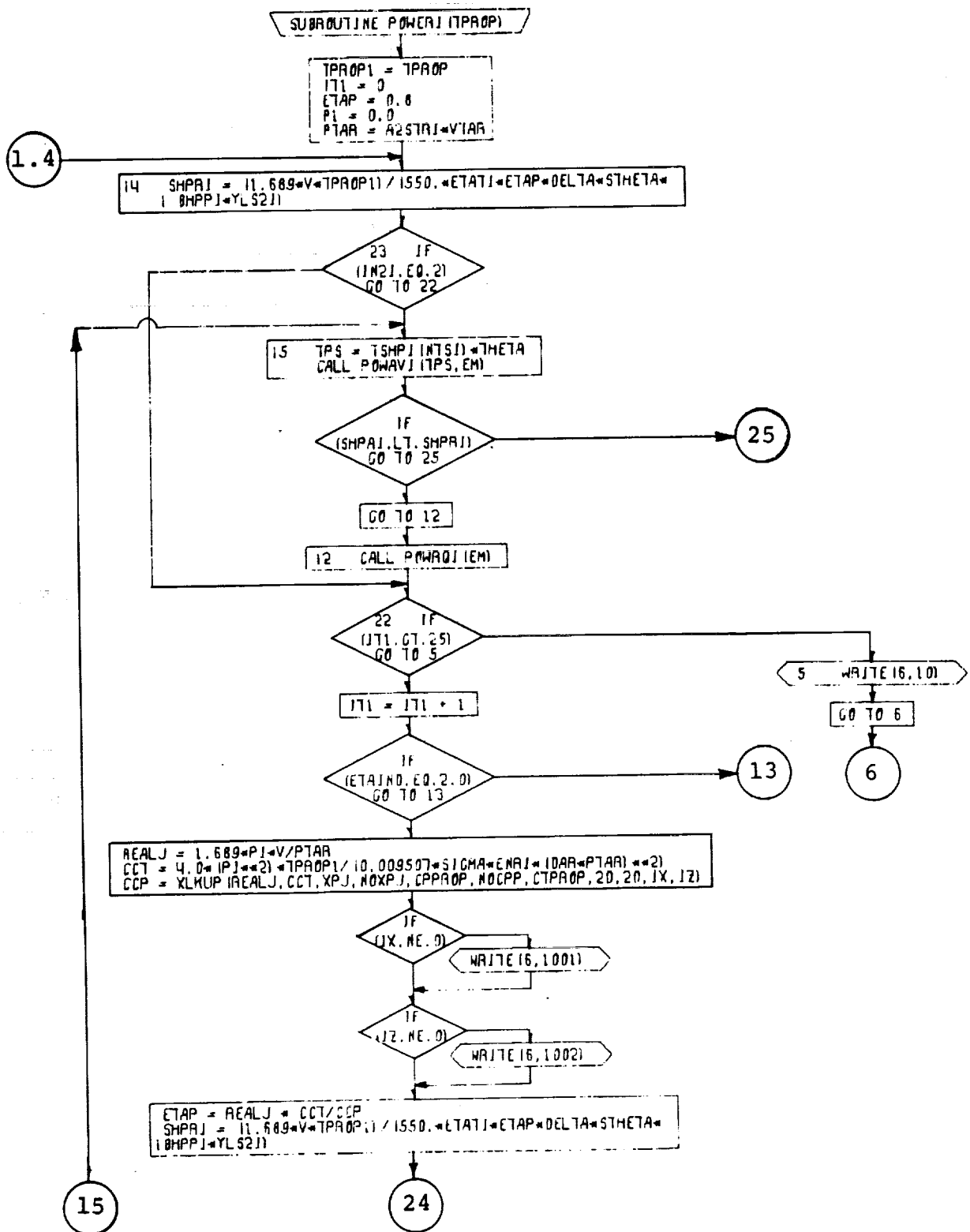


Figure 4-19. POWERI Subroutine Flow Chart (Part 1 of 3).

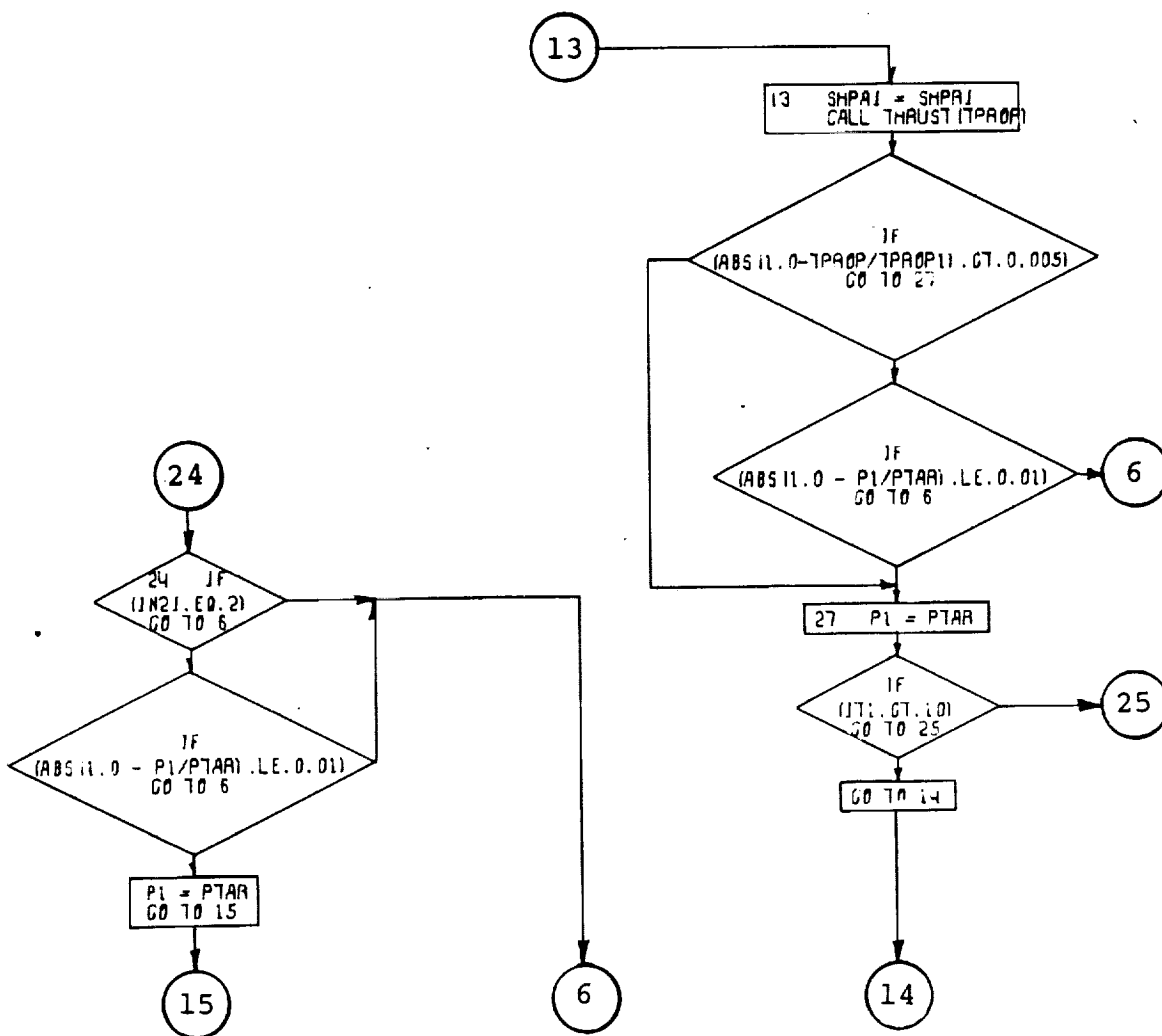


Figure 4-19. POWERI Subroutine Flow Chart (Part 2 of 3).

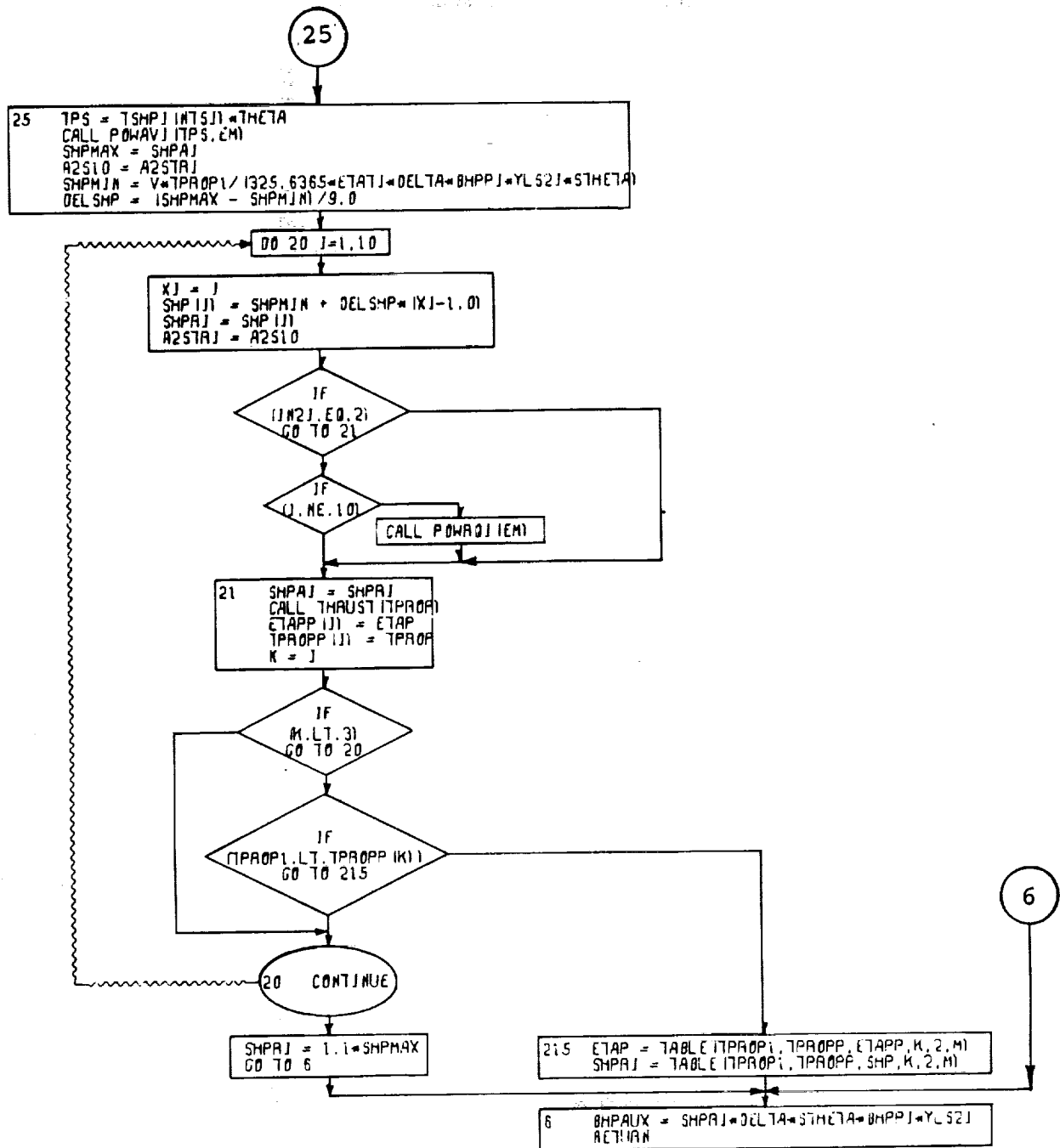


Figure 4-19. POWERI Subroutine Flow Chart (Part 3 of 3).

TABLE 4-3. COEFFICIENTS FOR PROPELLER EQUIVALENT POLARS

COEFFICIENTS FOR CRUISE:

C_L	M_H	a_0	a_1	a_2	M_H	a_0	a_1	a_2	M_H	a_0	a_1	a_2
0	0.7	90.	0.	0.	0.8	90.	0	0	0.9	90.	0.	0.
.05		7.0392	1.9949	61.2416		10.2148	2.4433	86.4731		11.3227	40.9515	21.9481
.10		4.8350	-4.1639	19.2195		8.3106	-22.6338	56.0		11.3355	-14.062	62.0142
.15		3.2218	-1.7030	8.291		5.4623	-14.9997	35.3636		8.3676	-12.2425	36.0889
.20		2.7551	-2.5322	7.0366		4.0458	-9.9837	23.0606		6.5856	-9.574	25.9573
.3		2.481	-4.5422	7.3774		3.9439	-13.0524	22.0028		5.3862	-8.9808	18.6439
.4		2.4521	-5.4949	7.4251		3.6769	-11.7146	17.3803		5.2054	-9.3153	16.4063
.5		2.8149	-7.092	8.3401		3.8766	-12.0044	16.0882		6.1902	-14.7567	23.2672
.6		3.8725	-10.861	11.4678		4.5901	-13.8756	17.2451		8.153	-25.0375	39.117
.7		5.6653	-16.2691	15.8093		6.1044	-18.2607	21.8349		10.1745	-30.7342	51.0509
.8		8.5799	-24.8115	22.6773		8.9031	-26.0958	30.7056		13.0822	-33.2211	58.1494
.9		12.25	-33.6185	28.7271		12.2042	-29.4588	34.1515		16.5344	-34.9378	64.3529
1.0		17.0496	-43.061	33.8798		17.0398	-37.3809	43.697		20.8089	-40.6314	76.3927
1.1		21.8332	-47.8821	33.6322		22.784	-47.3791	55.5455		25.6453	-49.145	94.4326
1.2		31.7062	-49.6246	26.4923		28.7851	-57.8217	68.2121		33.5049	-77.6449	144.4176

4.8 SIZE TRENDS SUBROUTINE

The size trends subroutine calculates the trends of the aircraft geometric dimensions as the weight of the aircraft changes throughout the iterative sizing loop. Figure 4-22 displays a flow chart showing the options available within the size trends subroutine.

The first of these is the option which determines main rotor diameter and solidity. It is possible to input diameter and solidity directly, or combinations of disc loading, design C_T/σ , diameter, and solidity. The following choices, specified by the main rotor sizing indicator, RDMIND, are available:

<u>RDMIND</u>	<u>INPUT</u>
1	Diameter and solidity
2	Disc loading and solidity
3	Diameter and C_T/σ
4	Disc loading and C_T/σ

If main rotor solidity is calculated, the program will choose the solidity satisfying the most critical of the three groups of requirements specified by input locations 0182 - 0190. These solidity sizing requirements are:

- (a) Solidity sized for hover conditions (Input $(C_T/\sigma)_H$, T/W)
- (b) Solidity sized for maneuver conditions (Input cruise speed, atmospheric conditions, maneuver C_T/σ , and rotor g loading)
- (c) Solidity sized for cruise conditions (Input cruise speed, atmospheric conditions, cruise C_T/σ , and rotor loading (N))

If so desired, the user may dictate which of these solidity choices the program makes simply by manipulating the inputs. For example:

If the solidity
sizing choice
desired is:

Then input:

Hover

Desired value for $(C_T/\sigma)_H$, $(C_T/\sigma)_{CR} =$
1.0, $g_{ROTOR} = .001$, $N(\text{Rotor Loading}) =$
0.1

Maneuver	$(C_T/\sigma)_H = 1.0$, Desired values for $(C_T/\sigma)_{CR}$ and g_{ROTOR} , N (Rotor Loading) = 0.1
Cruise	$(C_T/\sigma)_H = 1.0$, Desired value of $(C_T/\sigma)_{CR}$, $g_{ROTOR} = 0.001$, Desired value of N (Rotor Loading)

As noted earlier, two basic types, the single and tandem rotor helicopter, can be sized using this program. The following (beginning with the single rotor helicopter) provides a brief description of the options available to the user.

Tail rotor diameter may be input directly, or calculated. The choices open to the user are:

TRDIND

- 1 Tail rotor diameter calculated using a trend
- 2 Tail rotor diameter input directly
- 3 Tail rotor diameter calculated based on an input tail rotor disc loading

The tail rotor diameter trend used when TRDIND = 1 is illustrated in Figure 4-20 (see also Reference 6). The tail rotor disc loading input when TRDIND = 3 does not include vertical fin sideload losses.

Tail rotor solidity may be input directly or calculated. If calculated (TRDIND = 2), the tail rotor solidity is determined by either hover-antitorque requirements or hovering-turn requirements (including tail rotor precession effects, see References 5 and 6). The former is obtained by setting $\dot{\psi}$ (yaw rate) and $\ddot{\psi}$ (yaw acceleration) equal to zero.

Yaw moment inertia (I_{ZZ}) is required in calculating the tail rotor solidity for the single rotor helicopter in a hovering turn. The following equation is included in the size trends subroutine to determine the aircraft yaw inertia.

$$I_{ZZ} = \frac{W}{32.2} \left(0.115 K_{ZZZ} e \right)^2$$

Where I_{ZZ} = Yaw moment of inertia, slug ft²

W = Aircraft design gross weight, lb

K_{ZZZ} = Inertia adjusting factor (nominally = 1.0)

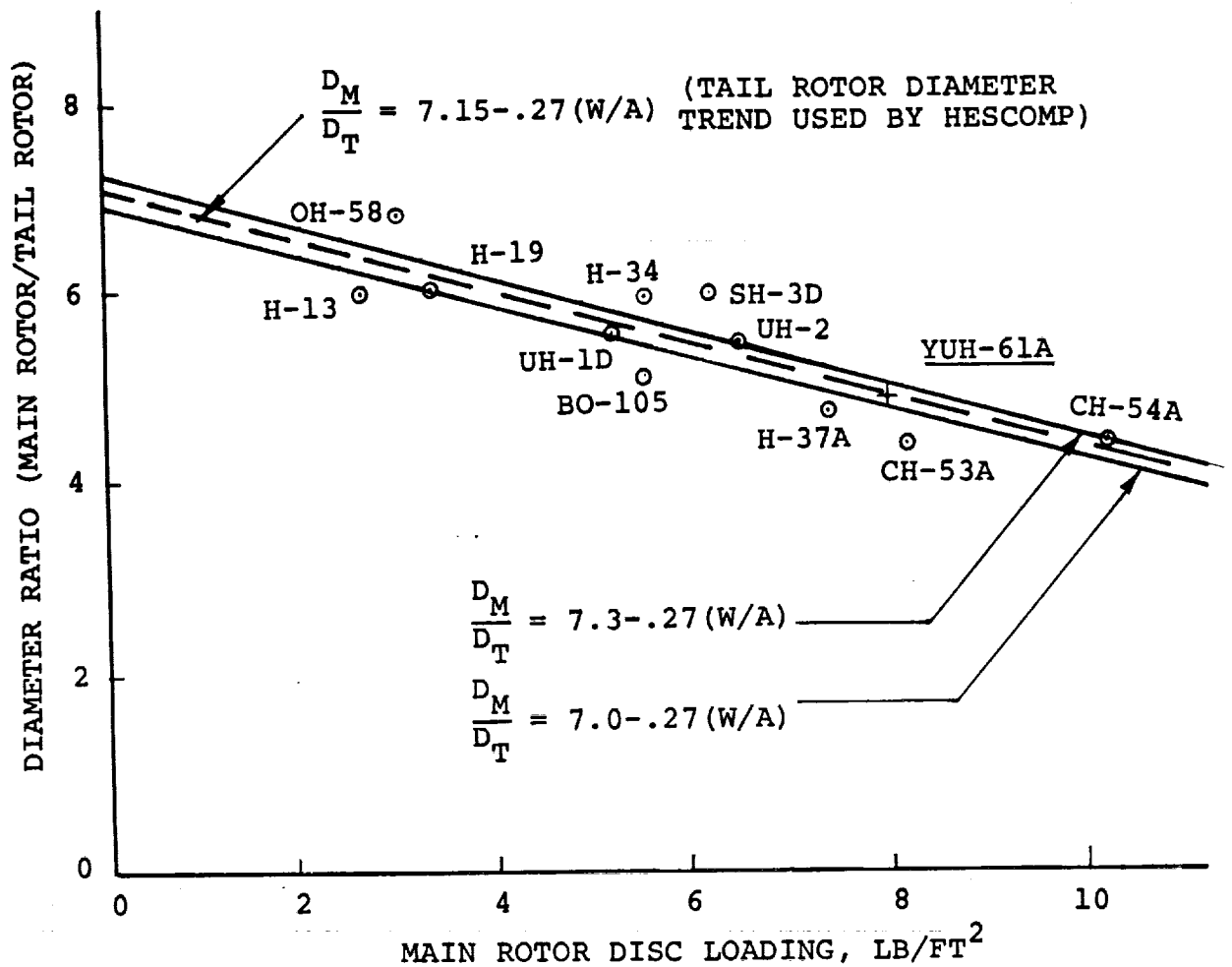


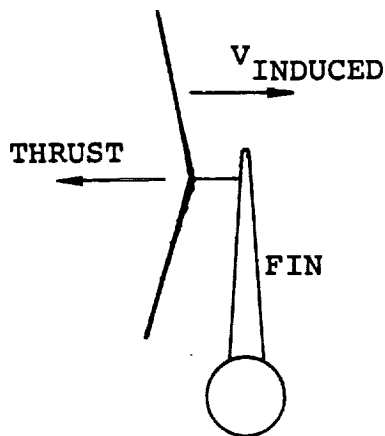
Figure 4-20. Tail Rotor Diameter Sizing Trend.

e = The combined sum of the study aircraft fuselage length and the cabin length measured from the nose of the aircraft to the end of the cabin.

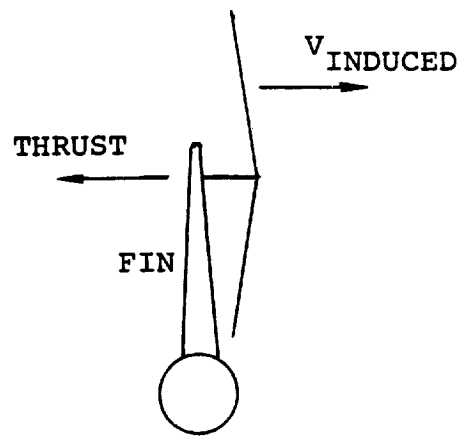
0.115 = Trend constant for determining the single rotor helicopter yaw moments of inertia.

To modify the equation inertia value, enter a fractional input in the K_{zzz} block (LOC 0213) (entering 1.1 will increase the 0.115 constant by 10 percent, entering 0.9 will decrease it by 10 percent, etc.)

It should be noted that the tail rotor gross/net thrust ratio (C_{TG}/C_{TNET}) may either be input directly or calculated. In the latter instance, C_{TG}/C_{TNET} is set equal to 1.00 and a value of the induced velocity ratio ($\bar{C}$) is input. Figure 4-21 illustrates typical values of $\bar{C}$ for both tractor and pusher tail rotor (see sketch below).



Tractor Tail Rotor



Pusher Tail Rotor

Note the difference in variations of $\bar{C}$ for the two different configurations. At low tail fin/rotor separation distances, the "tractor" values of $\bar{C}$ are sensitive to variations in tail rotor C_T . Thus, the closer the tractor tail rotor is located to the fin, the larger the error (admittedly small to begin with) involved in calculating C_{TG}/C_{TN} , since the user must "guess" what tail rotor C_T to use in selecting $\bar{C}$. The "pusher" $\bar{C}$ on the other hand is a function only of the fin/tail rotor separation distance.

In any event, use of this option is desirable in that tail rotor/fin sideload losses are matched to the vertical tail area calculated in the sizing process. Detailed explanations of all the factors involved in tail rotor design and sizing are contained in References 5 and 6.

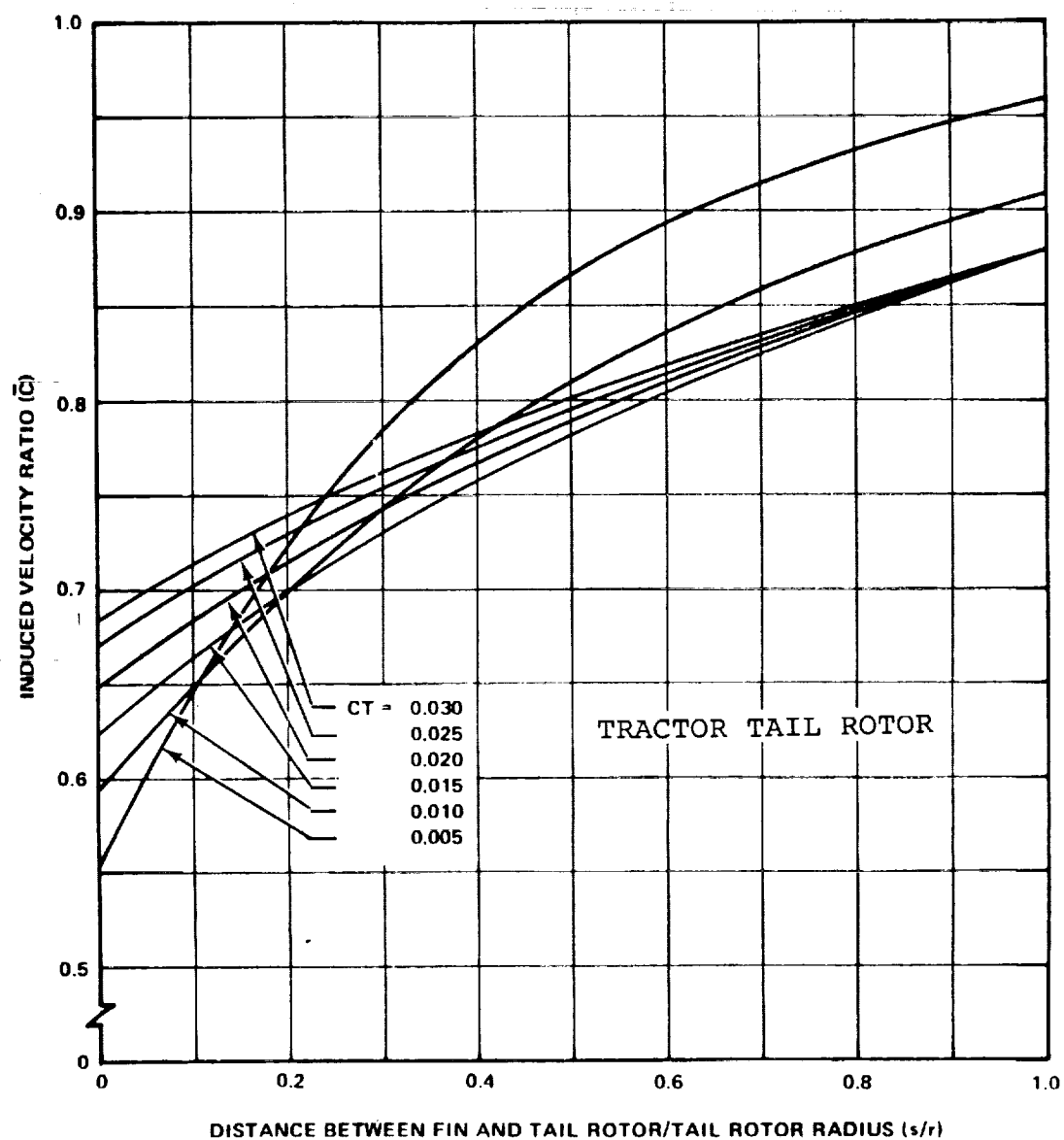


Figure 4-21. Tail Rotor/Vertical Tail Fin Interference Data (Part 1 of 2).

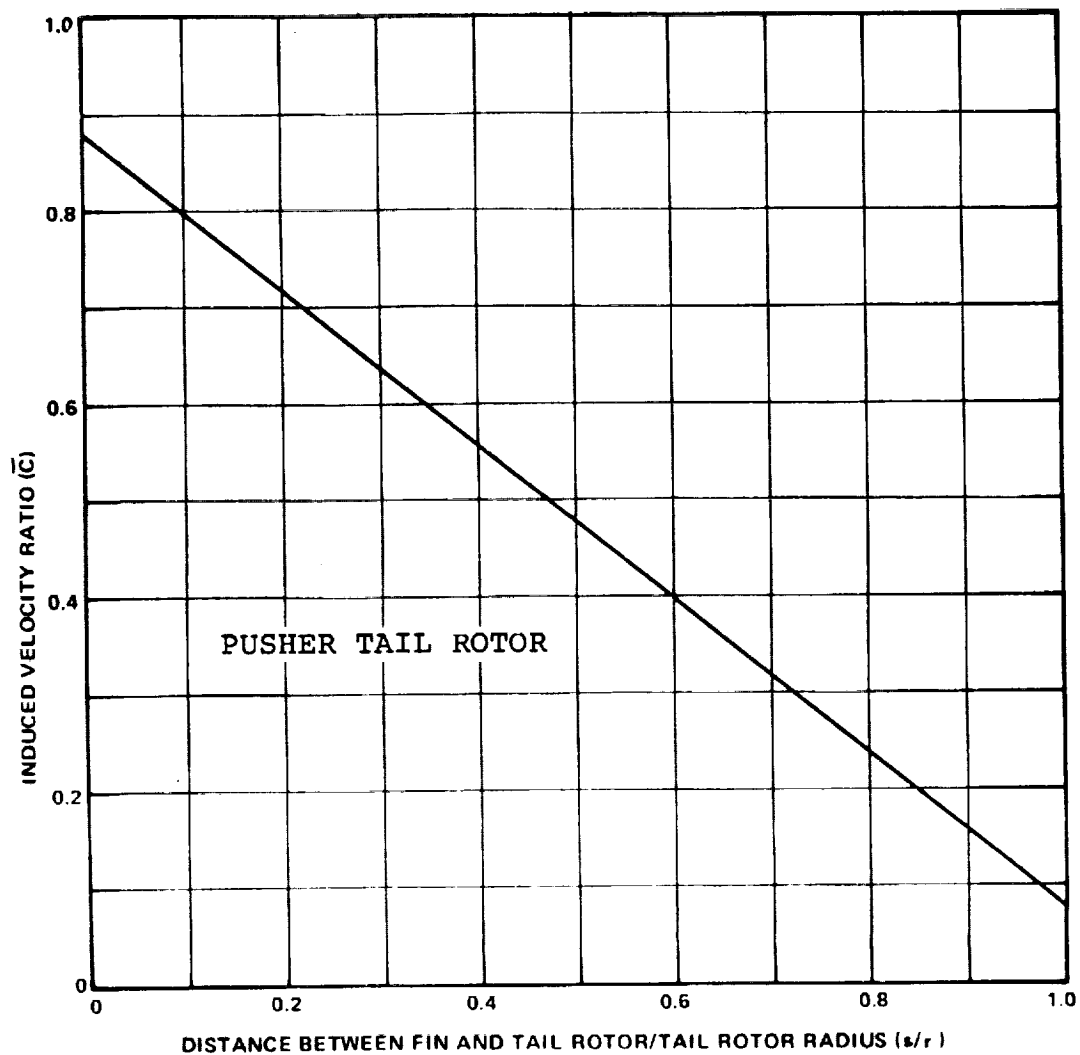


Figure 4-21. Tail Rotor/(Pusher Tail Rotor) Vertical Tail Fin Interference Data (Part 2 of 2).

The vertical tail size may be determined in three ways. If VTFIND = 1, aspect ratio and tail fin/tail rotor overlap is input. If VTFIND = 2, tail fin/tail rotor overlap and configuration directional stability requirements are input. If VTFIND = 3, the input is the same as with VTFIND = 2, with the exception that AR_{VT} is specified instead of tail rotor/fin overlap. These latter two options are important in that they allow the user to size the vertical tail to meet cruise anti-torque requirements at specified conditions (C_{LDes}, V_{Des}) in the event of tail rotor loss. It should be noted that C_{LDes} is assumed to represent the total lift coefficient developed by a conventional tail fin in sideslip, or a tail fin with a variable camber device (i.e., a rudder or flap) deployable under these circumstances.

Horizontal tail area is either directly specified (HTIND = 1) or calculated (HTIND = 2). In the latter case, tail area is calculated based on a tail volume coefficient specified by the following equation:

$$V_{HT} = \frac{16 \ell_{TH} S_{HT}}{\pi^2 D_{MR}^3}$$

where: ℓ_{TH} = distance from rotor center to 1/4 chord of horizontal tail

D_{MR} = main rotor diameter

S_{HT} = horizontal tail planform area

Since the horizontal tail is used to offset longitudinal trim changes caused by the main rotor, it is important to use a tail volume coefficient that reflects the type of rotor system in use. For example, in sizing a helicopter with a hingeless rotor system, the tail volume coefficient (V_{HT}) would be obtained by taking an existing hingeless rotor helicopter, measuring the applicable dimensions from a drawing and calculating V_{HT} .

Forward rotor pylon dimensions are specified directly (input LOCS 0152 - 0156).

The computer program calculates the length and wetted area of the fuselage based upon input values of cabin length, cabin mean diameter, fineness ratios of the pilots section and tail section, and calculated tail boom dimensions. The tail boom length is established by the tail rotor diameter, the need to maintain a reasonable gap between the main and tail rotor discs and the relative position of the main rotor on the fuselage. This position (X_M/ℓ_B) may either be input

(MRPIND = 0) or calculated (MRPIND = 1, 2), using a simple weight-balance subroutine. If the latter option is chosen, the relative positions (from the aircraft nose) of the various aircraft components (engines, primary drive system, etc.), must be input (LOCS 2678-2696). Additional increments of fuselage wetted area (to account for miscellaneous bulges, fairings, etc.) may be input through the use of $\Delta S_{Wet}/S_F$ (LOC 0120) and ΔS_{Wet} (LOC 0121).

Three options are available for sizing a tandem rotor helicopter fuselage. These options, specified by the indicator FDMIND are:

<u>FDMIND</u>	<u>Input</u>	<u>Calculated</u>
1	((O/L)/D), forward, aft rotor positions	fuselage length (l_F) cabin length (l_C)
2	((O/L)/D), cabin length (l_C)	fuselage length (l_F) forward & aft rotor positions
3	cabin length (l_C), forward and aft rotor positions	((O/L)/D), fuselage length (l_F)

In cases (FDMIND = 1) where the calculated cabin length is less than zero, an error statement is printed and the case terminated. Likewise, if the rotor overlap/diameter ratio exceeds either +0.5 or -0.5, the case is terminated.

The aft rotor pylon dimensions may either be input directly (APHIND = 1) or calculated (APHIND = 2) based on an input of rotor gap/stagger ratio. The forward pylon dimensions are input directly as in the case of the single rotor helicopter.

In the case of a compound helicopter, propeller dimensions and characteristics (i.e., AF, blade number, C_{L_i} , etc) are input directly.

Wing sizing options are divided into two groups, those for determining wing area (S_W IND) and those for determining wing span (b_W IND). Wing area may either be input directly (S_W IND = 1), calculated based on an input wing loading (S_W IND = 2), or sized to meet a maneuver requirement. In the latter case, the wing size is dictated by the need to carry the difference between the overall g requirement (LOC 0188) and the maneuver g's (LOC 0189) carried by the main rotor(s). Wing span may be determined on the basis of an input wing span/rotor diameter ratio (b_W IND = 1), an input aspect ratio (b_W IND = 2); or, in the case of a compound helicopter with wing mounted propellers, on the basis of propeller tip/fuselage clearance considerations (b_W IND = 3).

The dimensions of the primary and auxiliary independent engine nacelles are determined by the horsepower or thrust level of the engines. Separate input constants z_1 , z_2 , z_3 , z_4 , z_5 and z_6 are used to calculate the size of the nacelles.

Primary engine nacelles

$$\text{Diameter (ft)} = z_1 \left[\frac{\text{SHP}^*}{N_P} \right]^{1/2}$$

$$\text{length (ft)} = z_2 + z_3 \left[\frac{\text{SHP}^*}{N_P} \right]^{1/2}$$

$$\text{wetted area (ft}^2\text{)} = N_P \pi (\text{dia}) (\text{length})$$

Auxiliary independent engine nacelles

$$\text{diameter (ft)} = z_4 \left[\frac{\text{SHP}^*_i}{N_{P_i}} \right]^{1/2} \quad \text{or} \quad z_4 \left[\frac{\text{FN}^*}{N_{P_i}} \right]^{1/2}$$

$$\text{length (ft)} = z_5 + z_6 \left[\frac{\text{SHP}^*_i}{N_{P_i}} \right]^{1/2} \quad \text{or} \quad z_5 + z_6 \left[\frac{\text{FN}^*}{N_{P_i}} \right]^{1/2}$$

$$\text{wetted area (ft}^2\text{)} = N_{P_i} \pi (\text{dia}) (\text{length})$$

Figure 4-22 show a flow chart of this subroutine.

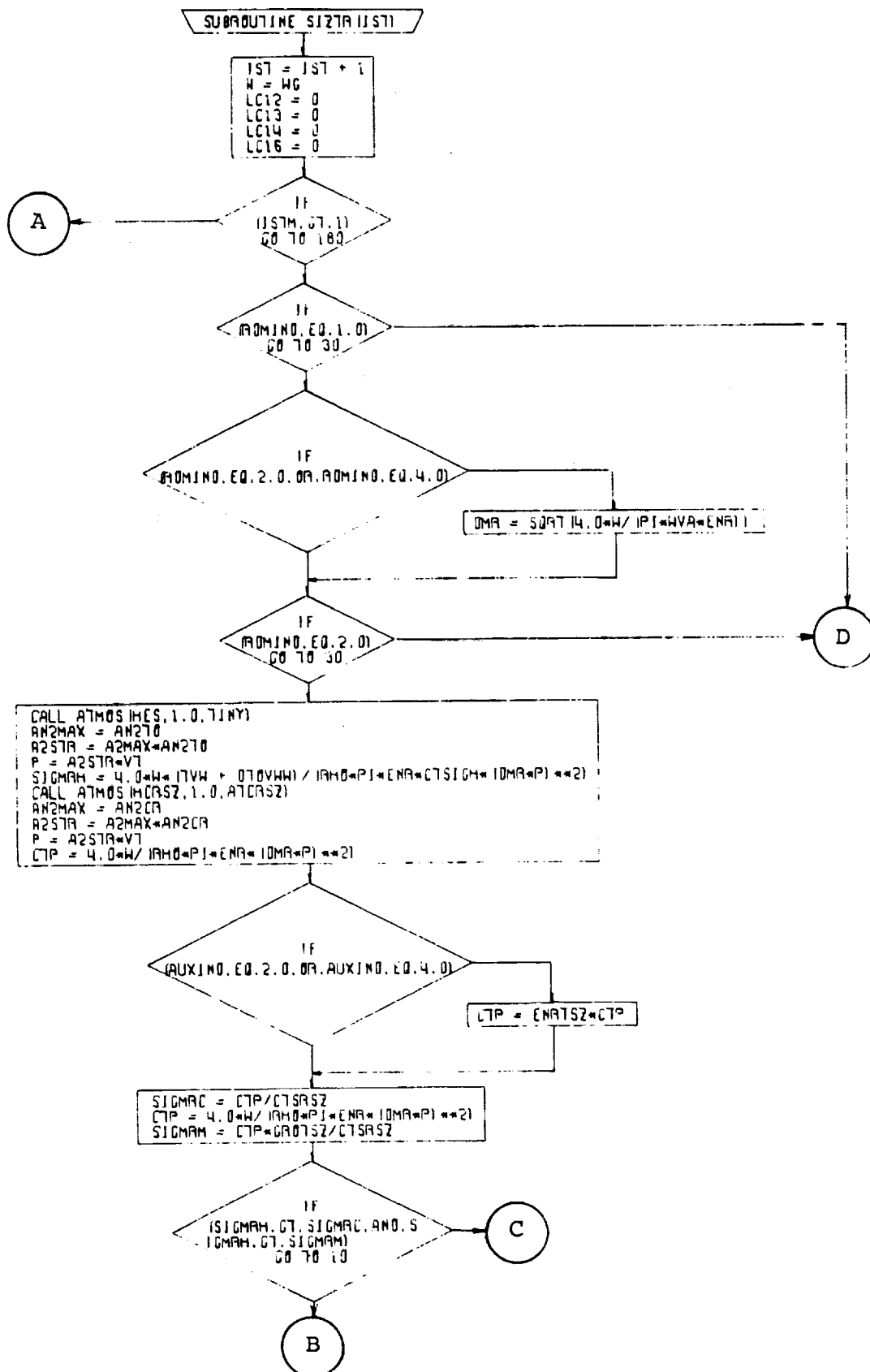


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart (Part 1 of 13).

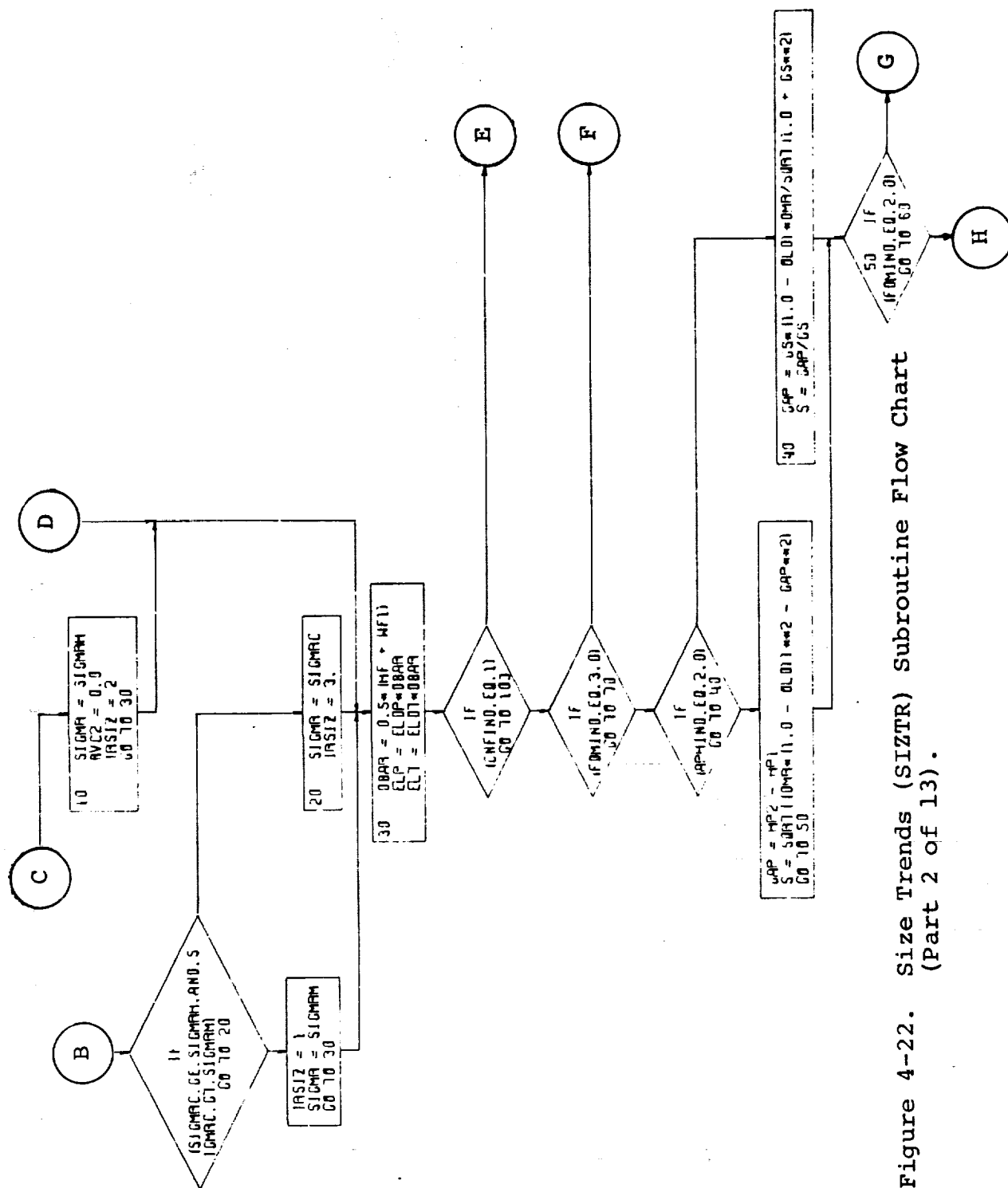


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart
(Part 2 of 13).

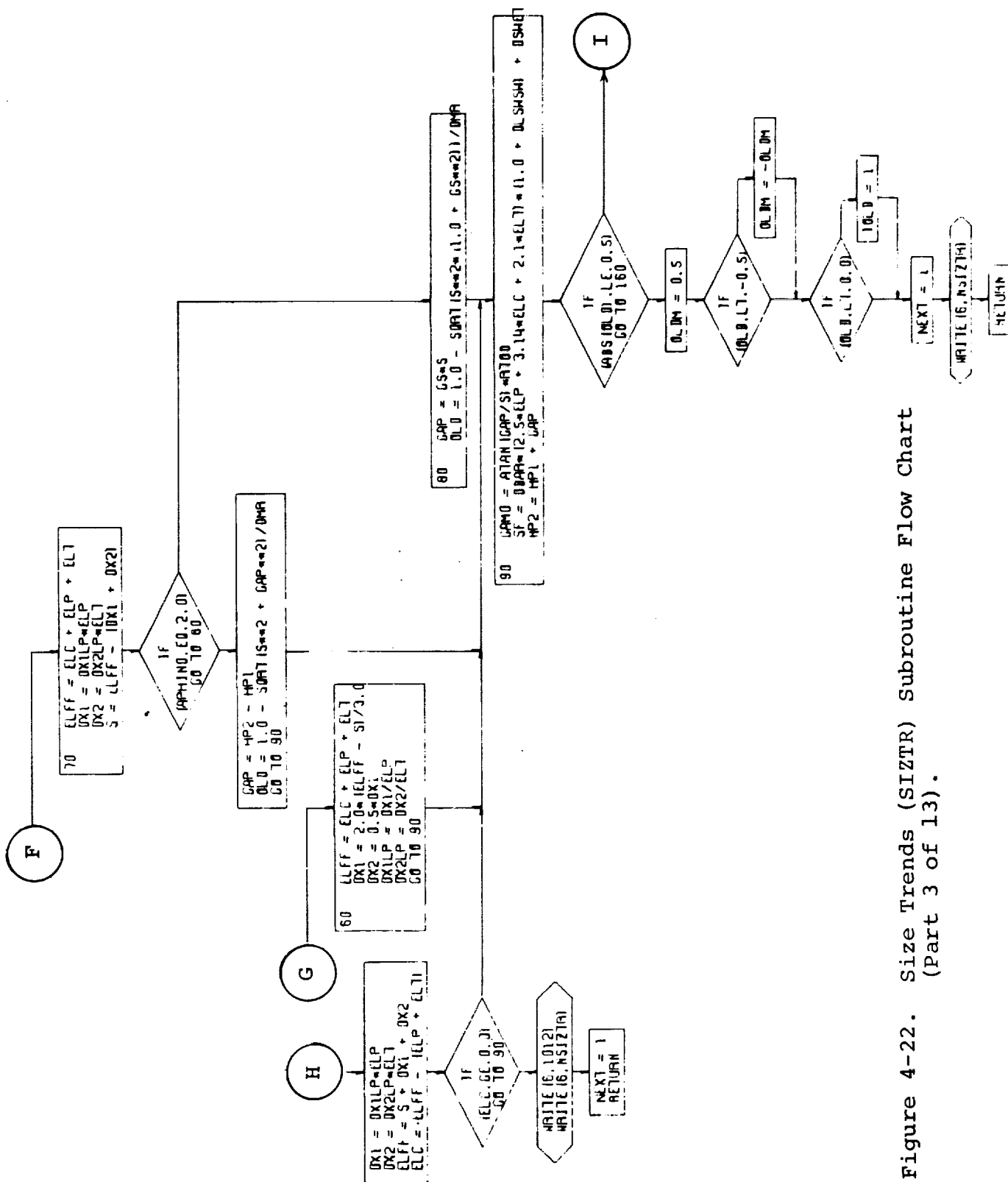


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart
(Part 3 of 13).

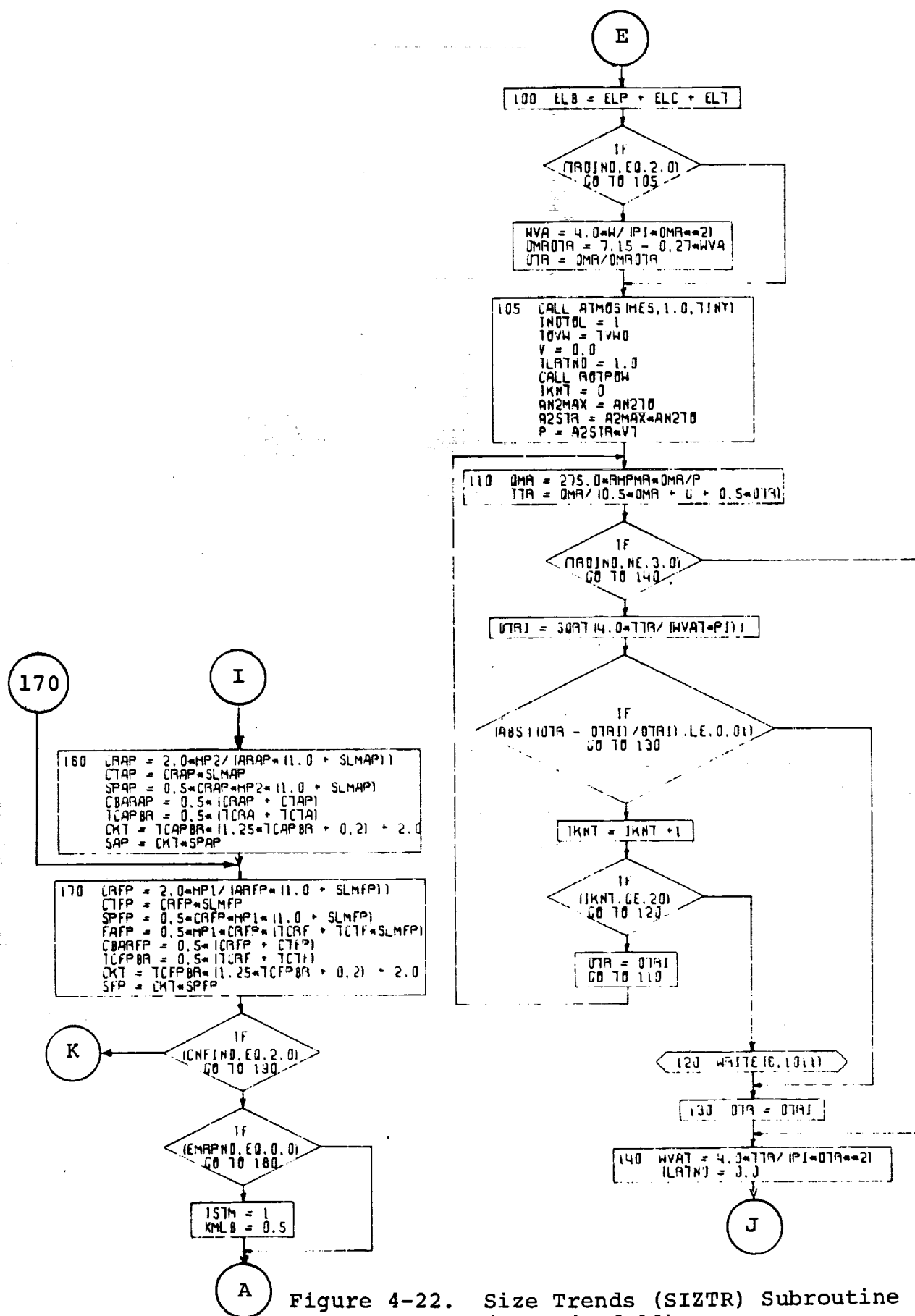
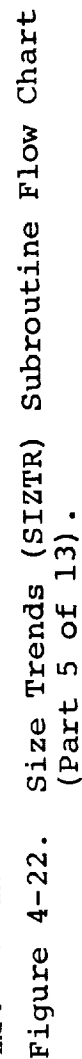


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Char
(Part 4 of 13).



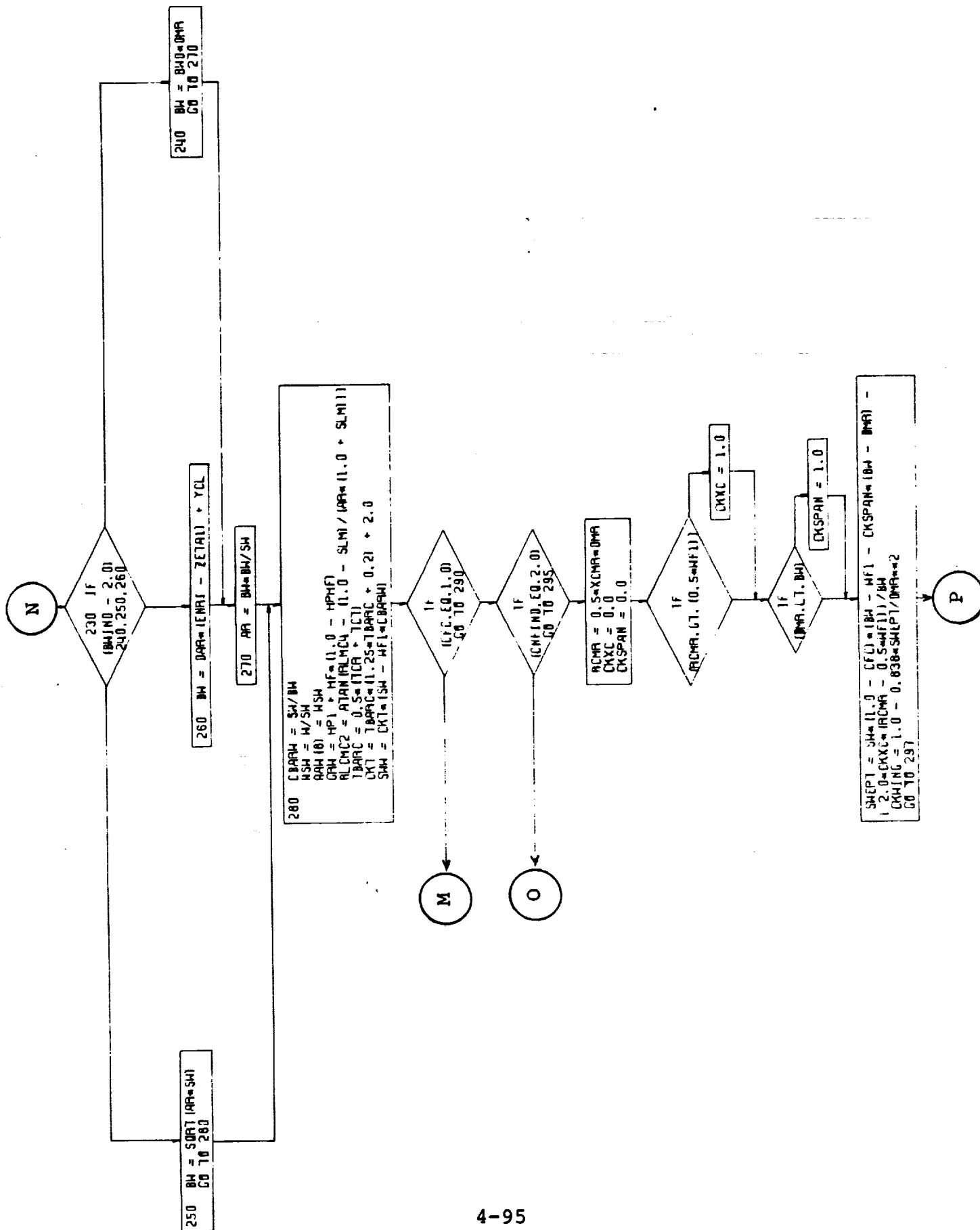


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart (Part 6 of 13).

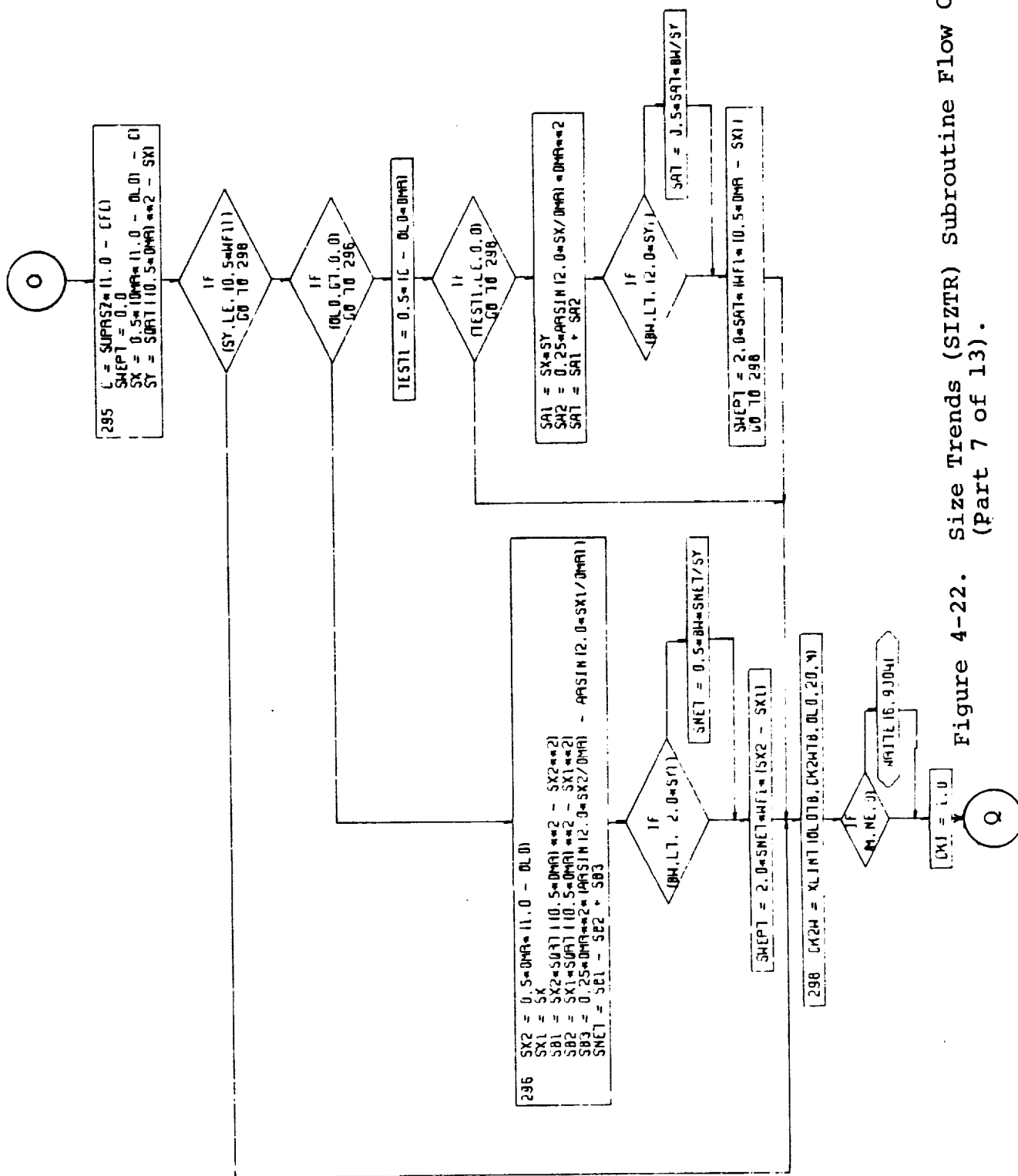


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart (Part 7 of 13).

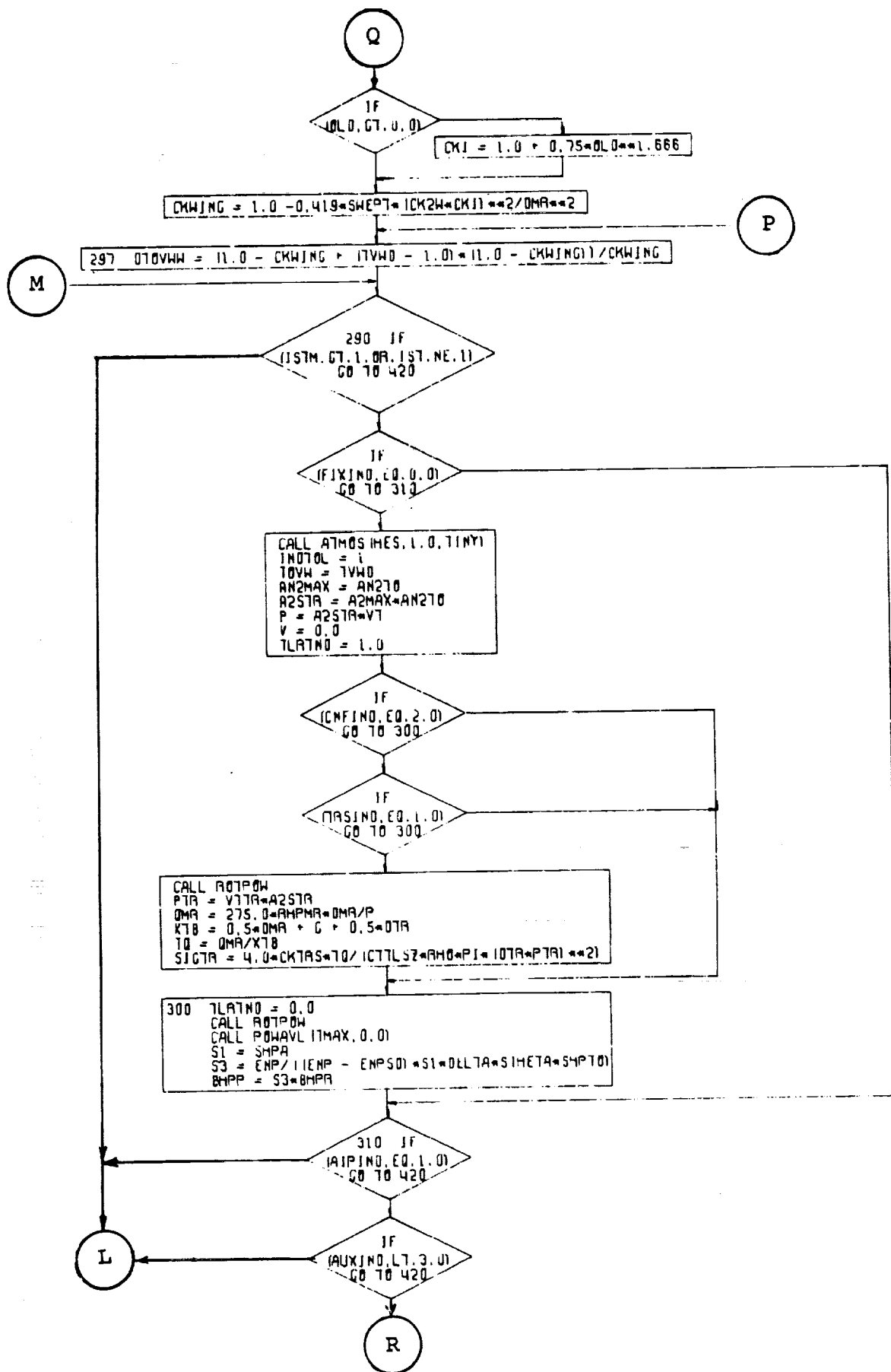


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart (Part 8 of 13).

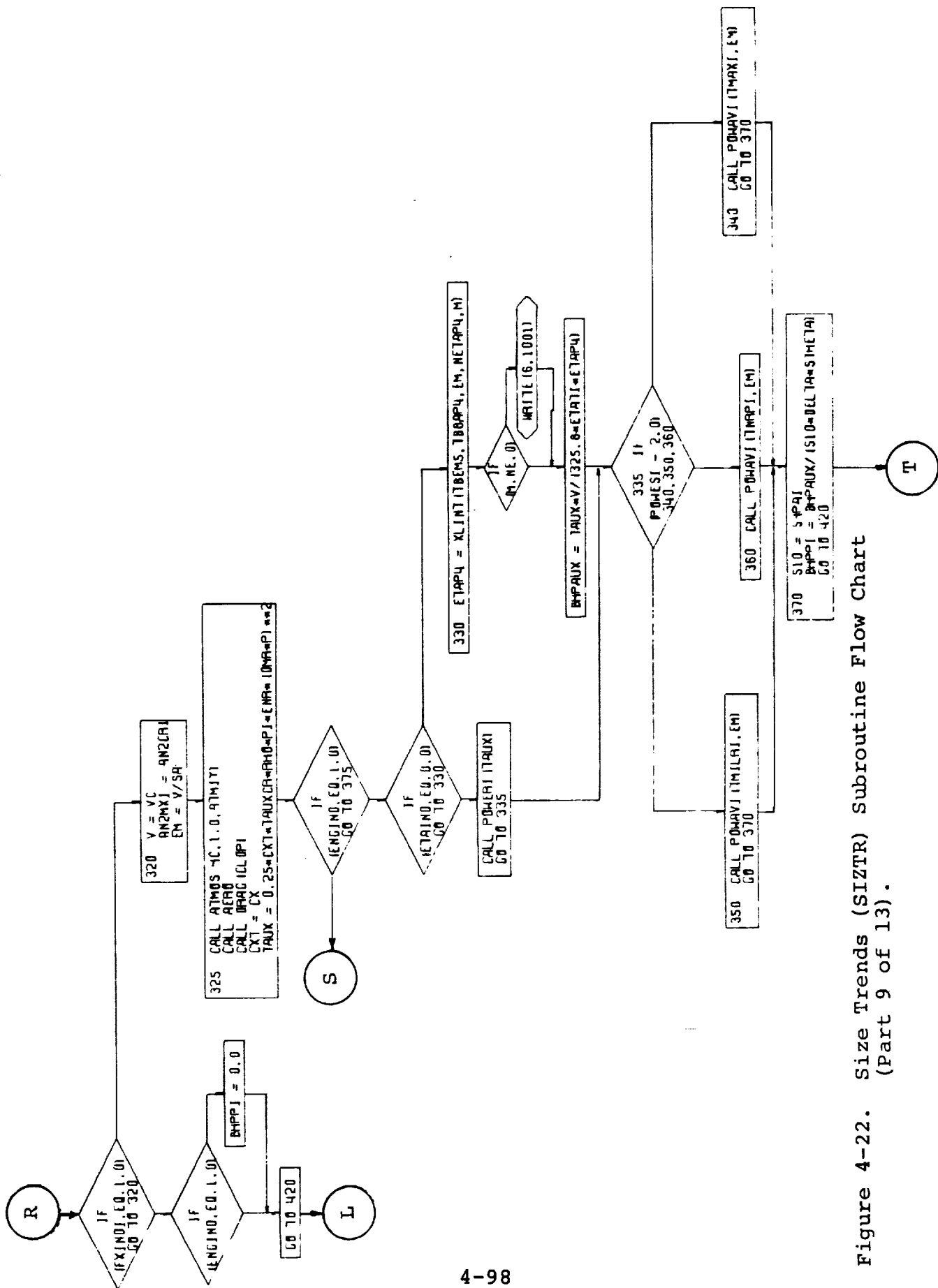


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart
(Part 9 of 13).

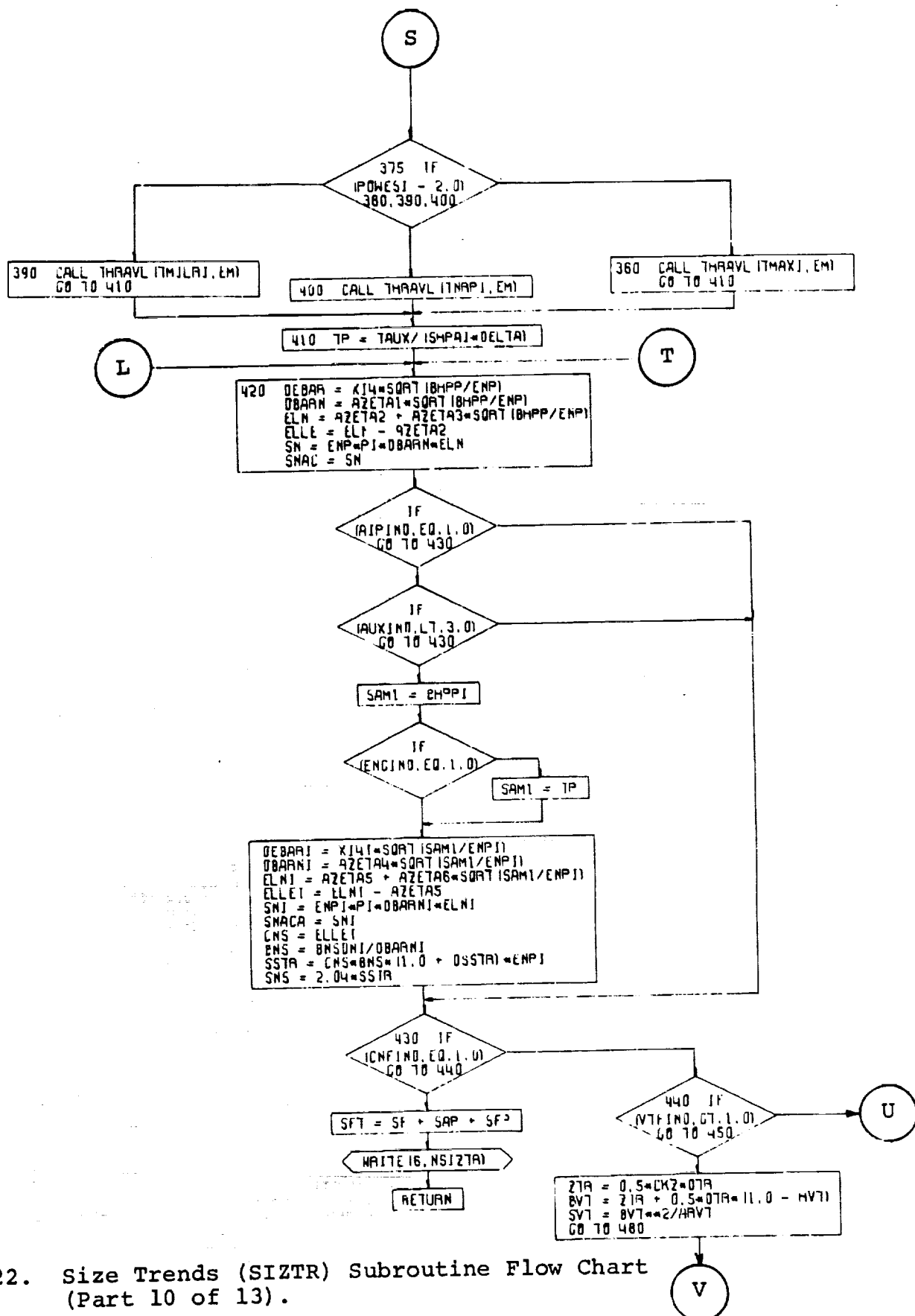


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart (Part 10 of 13).

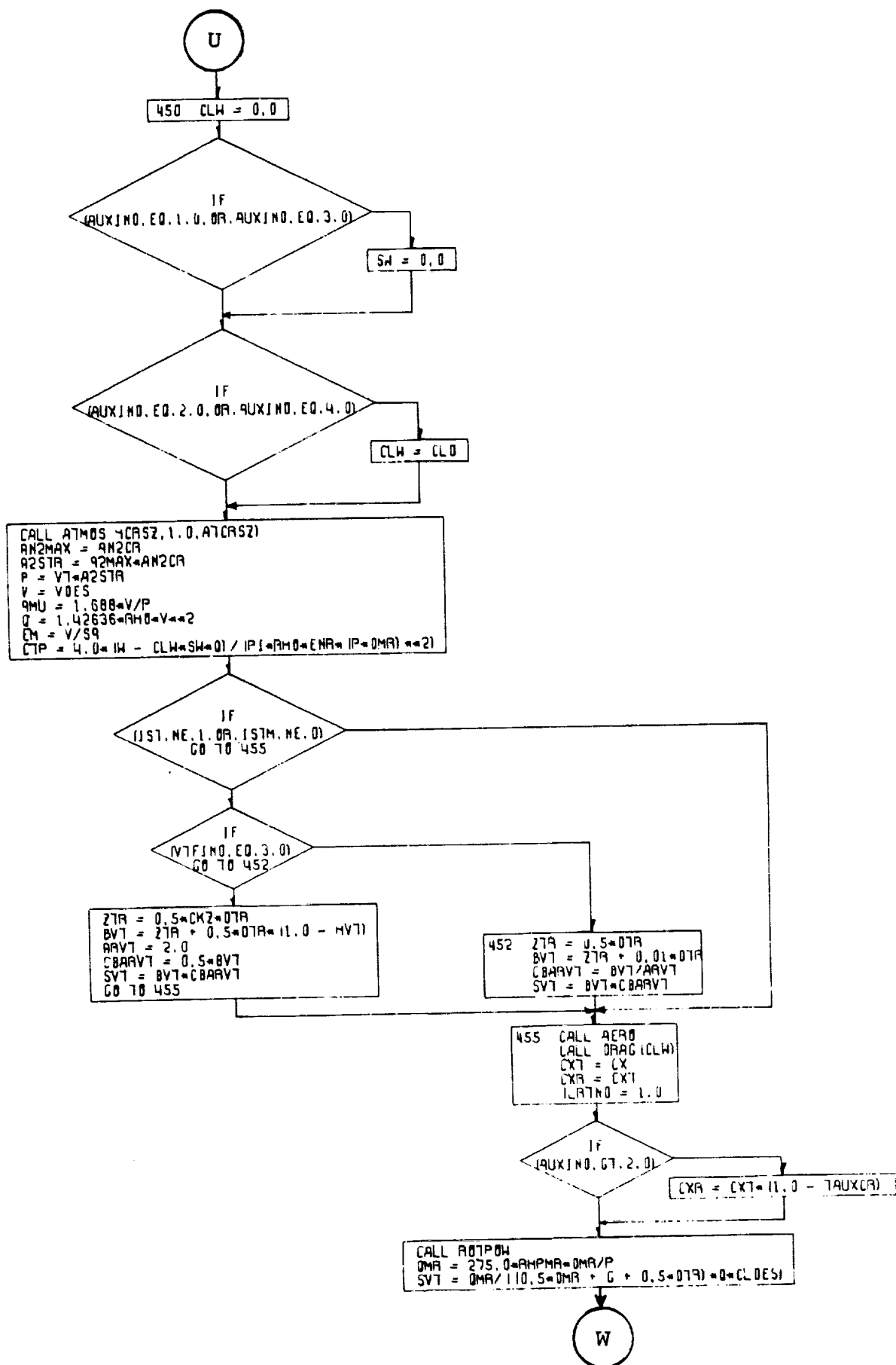


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart
(Part 11 of 13).

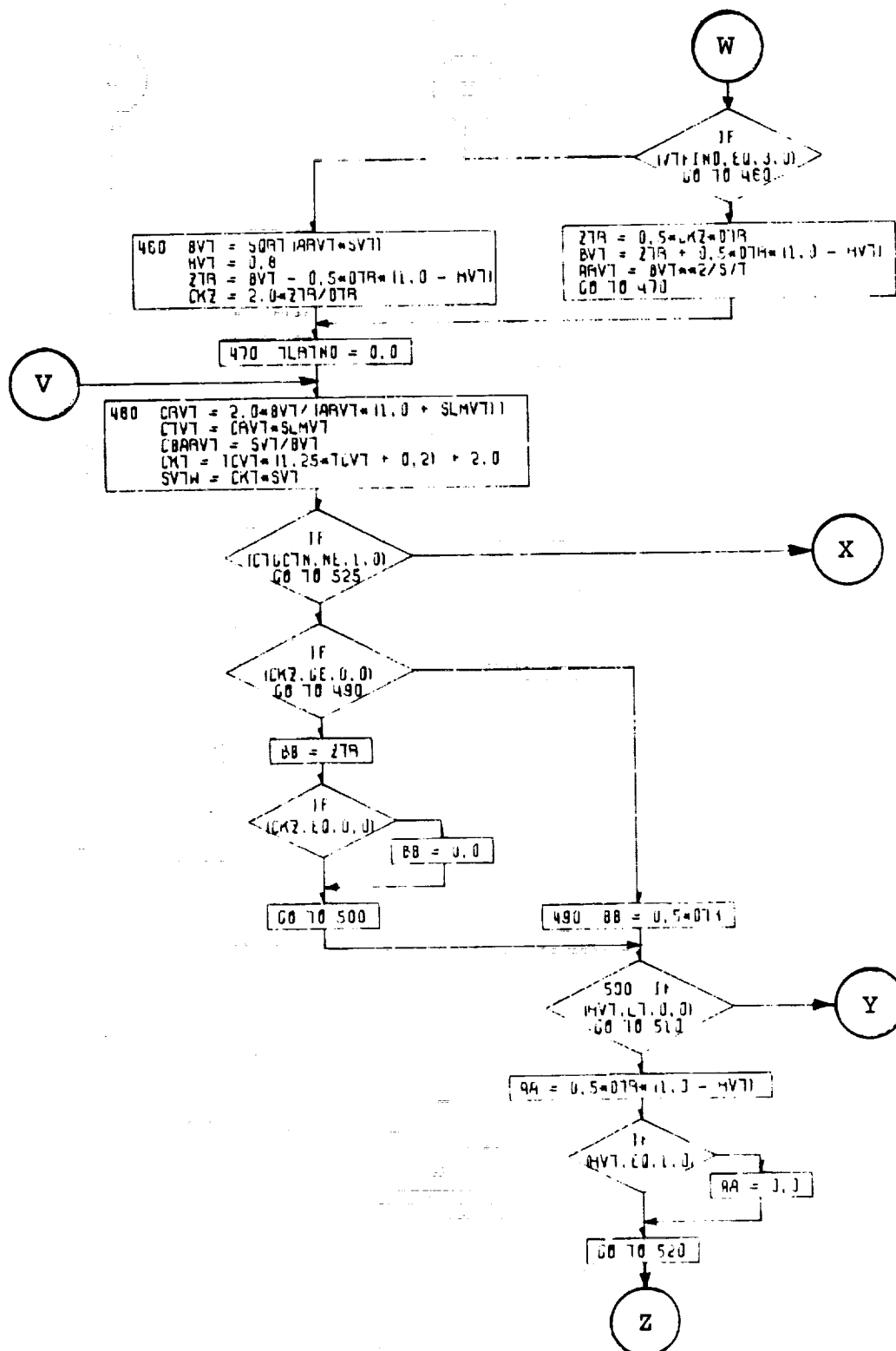


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart
(Part 12 of 13).

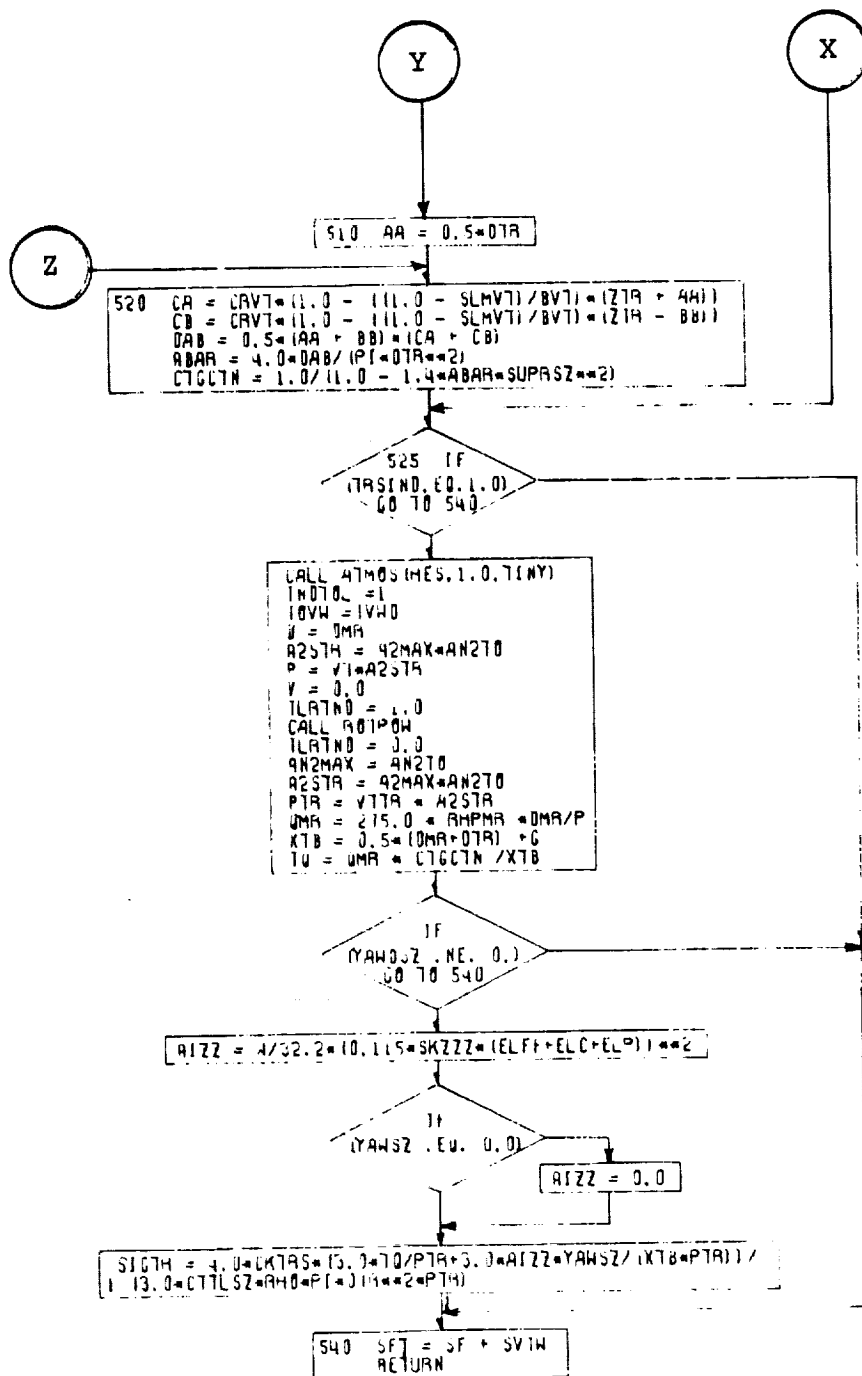


Figure 4-22. Size Trends (SIZTR) Subroutine Flow Chart
(Part 13 of 13).

4.9 AERODYNAMICS CALCULATIONS SUBROUTINE

The aerodynamics subroutine calculates a series of factors (a_5 , a_6 , a_7 , a_8 , and a_9) which are used in the calculation of drag. The drag calculation has been written in the most general manner possible. Drag is assumed to be divided into profile, induced, and interference components; namely,

$$F_{eTOT} = a_5 + \underbrace{a_6 C_{DWi} S_W}_{\text{Wing Profile Drag}} + \underbrace{a_7 C_{LW}^2 S_W}_{\text{Wing Induced Drag}} + \underbrace{a_9 C_{LFIN}^2 S_{VT}}_{\text{Vertical Tail Induced Drag}} + F_{eIF}$$

where,

F_{eTOT} = Total configuration equivalent flat plate drag area

a_5 = Basic configuration flat plate drag area (including fuselage, rotor hubs, rotor pylons, etc).

$a_6 = K_W [f_W (Re)]$

$a_7 = 1/\pi e AR$

$a_9 = 1/\pi e_{VT} AR_{VT} TFEF$

F_{eIF} = Rotor/wing interference equivalent flat plate drag area (calculated by the program using simplified Prandtl Bi-Plane Theory).

The basic configuration flat plate drag area may be calculated in two different ways: by a detailed build up, or by a trend (see Figure 4-23). The wing profile drag is assumed to be a function of lift coefficient, as specified by an input table.

If the user elects to determine the basic configuration flat plate drag area (a_5) by build up ($DRGIND = 1$), then profile drag coefficients (C_{DHT} , C_{DVT} , etc.) and form factors (K_{HT} , K_{VT} , etc.) for each component are input to the program, a_5 then being calculated from the following relationship:

$$a_5 = \underbrace{.00287 K_F S_F [f_F (Re)]}_{\text{Fuselage Profile Drag}} + \underbrace{K_{FP} C_{DFP} F_{AFP}}_{\text{Forward (Main) Rotor Pylon Profile Drag}}$$

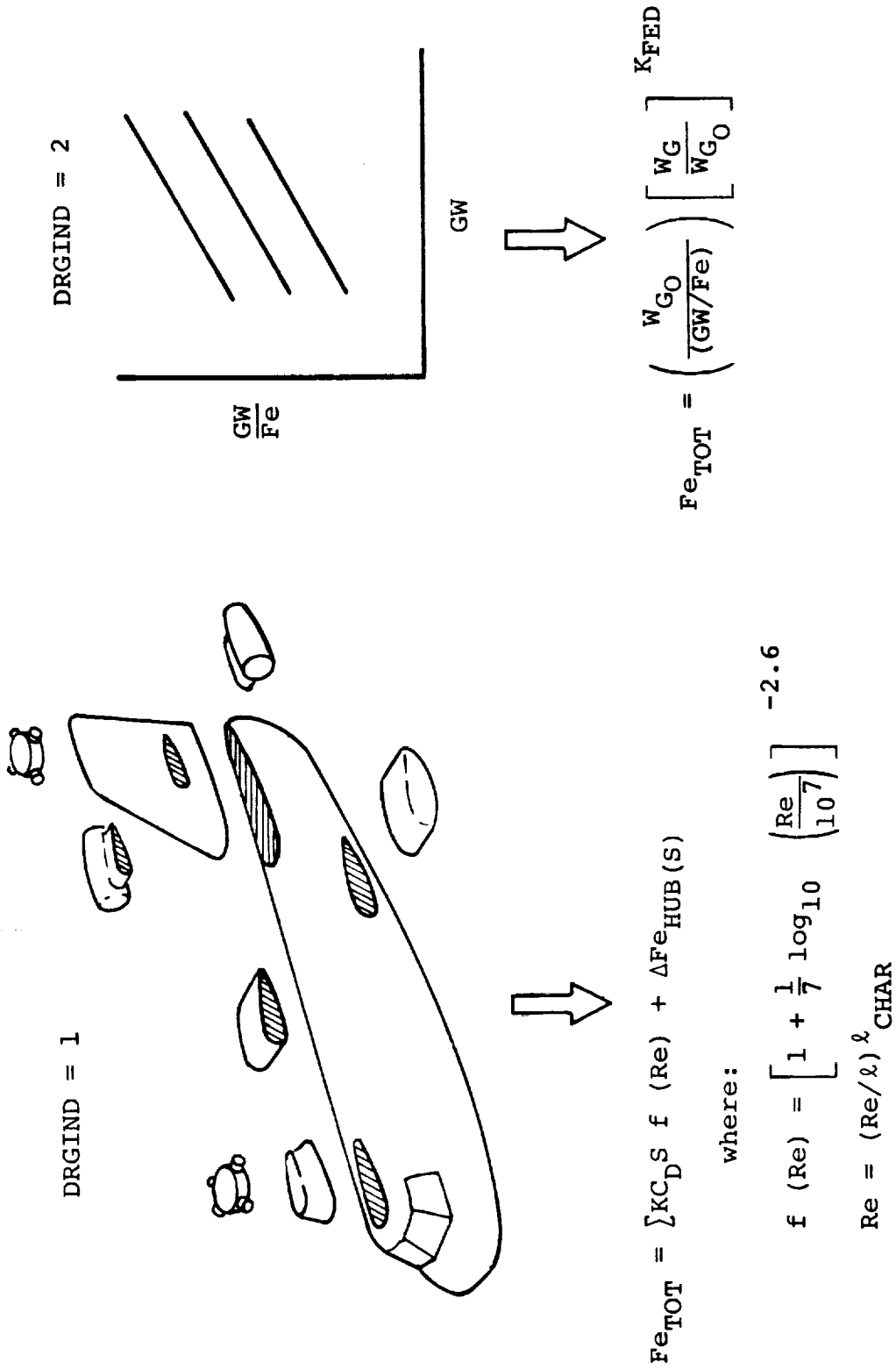


Figure 4-23. Parasite Drag Buildup Options.

$$+ \underbrace{K_{AP} C_{DAP} S_{AP} [f_{AP} (Re)]}_{\text{Aft Rotor Pylon Profile Drag}} + \underbrace{K_{VT} C_{DVT} S_{VT} [f_{VT} (Re)]}_{\text{Vertical Fin Profile Drag}}$$

$$+ \underbrace{K_{HT} C_{DHT} S_{HT} [f_{HT} (Re)]}_{\text{Horizontal Tail Profile Drag}} + \underbrace{K_N C_{DN} S_N [f_N (Re)]}_{\text{Primary Engine Nacelle(s) Profile Drag}}$$

$$+ \underbrace{K_{NI} C_{DNI} S_{NI} [f_{NI} (Re)]}_{\text{Auxiliary Independent Engine Nacelle Profile Drag}} + \underbrace{K_{NS} C_{DNS} S_{NS} [f_{NS} (Re)]}_{\text{Auxiliary Independent Engine Nacelle Strut Profile Drag}}$$

$$+ \underbrace{\Delta Fe_{MRH} N_R}_{\text{Main Rotor Hub(s) Total Drag}} + \underbrace{\Delta Fe_{TRH}}_{\text{Tail Rotor Hub Total Drag}} + \underbrace{\Delta Fe [f_F (Re)]}_{\text{Miscellaneous Drag}}$$

The terms $f_w (Re)$, $f_F (Re)$, $f_{VT} (Re)$, etc., are Reynolds number functions for the wing, fuselage, vertical tail, etc., which reflect the variation of skin friction coefficient with Reynolds number. The function which is used is a normalized form of the Prandtl-Schlichting turbulent flat plate skin friction equation:

$$f(Re) = \frac{C_f}{C_{fRe=10^7}} = [1 + \frac{1}{7} \log_{10} \frac{Re}{10^7}]^{-2.6}$$

The program user inputs a value for average Reynolds number per foot for the mission and the program then calculates the Reynolds' number for each component of the aircraft and uses the Reynolds' number functions $f_w (Re)$, $f_F (Re)$, etc., to determine the variation in component drag as the aircraft dimensions change during the iteration on gross weight. The individual profile drag coefficients, C_{DVTi} , C_{DHTi} , etc., are input at a reference Reynolds number of 10^7 .

Particular care must be exercised in the input of data required for calculating the hub drag, as this particular component can typically account for as much as 1/2 to 2/3 of the

total parasite drag of a helicopter. The hub drag calculation method (Reference 8) used in this program is based on the following relationships:

$$\Delta Fe_{HUB} = \underbrace{Fe_{HUB_{CS}}}_{\text{Hub center}} + \underbrace{Fe_{SH}}_{\text{Rotor shanks}} + \underbrace{Fe_{INT}}_{\text{Misc hub/shank Interference}}$$

where:

$$Fe_{HUB_{CS}} = f \left\{ C_{D_{CS}} \text{ (Hub projected frontal area)} \right\}$$

$$Fe_{SH} = f \left\{ C_{D_{SH}}, \text{ Number of shanks, shank projected frontal area, and local advance ratios at the hub/shank and shank/rotor blade interfaces} \right\}$$

Typical values of hub center section and shank drag coefficients are illustrated in Figure 4-24. A few notes of caution on the use of values such as are contained in Figure 4-24 is in order. First, estimates of the Reynolds number of the hub or shank sections to be used should be made in order to establish whether the sections are sub or super critical. Second, while 2-dimensional section drag coefficients are appropriate for use with shanks of extended length, test data indicates that 3-dimensional coefficients are more representative for low aspect ratio shanks bounded by lower drag shapes (for example, a short stubby shank bounded on one side by a faired hub and on the other side by the root end of the rotor blade). Figure 4-25 illustrates the hub geometry and interference drag factors implicitly assumed in this program. If the actual hub geometry or interference drag variation desired by the user differs appreciably from these, the differences can be reflected by ratioing the input drag coefficients $C_{D_{CSMR}}$, $C_{D_{SHMR}}$, $C_{D_{CSTR}}$, and $C_{D_{SHTR}}$ accordingly. Table 4-4 summarizes these corrections.

If it is desired to calculate a_5 by use of a drag trend (DRGIND = 2), the following relationship then obtains:

$$a_5 = \left[\frac{W_{G0}}{(G_W/Fe)} \right] \left[\frac{W_G}{W_{G0}} \right]^{K_{FED}}$$

INFLUENCE OF CROSS SECTION SHAPE ON SHANK DRAG COEFFICIENTS (CYLINDERS AT SUPERCRITICAL REYNOLDS NO.)

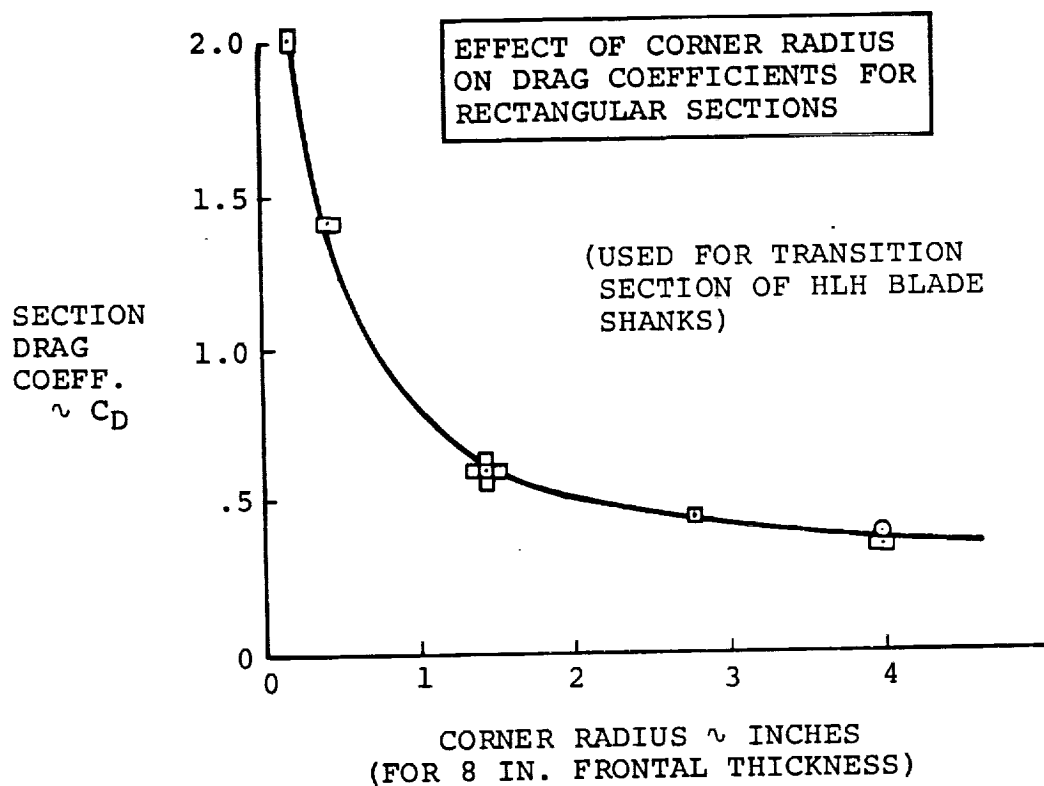
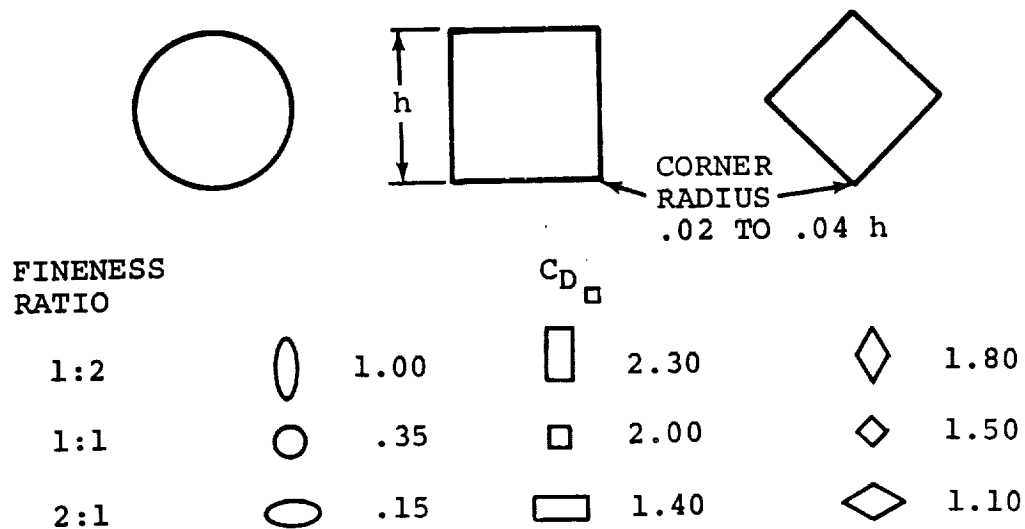


Figure 4-24. Typical Hub and Shank Drag Coefficients (Part 1 of 2).

HUB CENTERSECTION DRAG COEFFICIENTS

CH-47 HUB (INCLUDING INTERFERENCE)

@ $\alpha_{\text{FUSE}} = 0$ $C_D = 1.61$

@ $\alpha_{\text{SHAFT}} = 0$ $C_D = 1.91$

CH-47 HUB (WITH INTERFERENCE CORRECTION)

BASED ON STATIC AREA $C_D = 1.03$

BASED ON ROTATING AREA $C_D = .88$

DRTS/12' DIA. - 3/6 BLADED ROTOR HUB (NO INTERFERENCE)

BASED ON STATIC AREA $C_D = 1.03$

BASED ON ROTATING AREA $C_D = .88$

NASA MEMO 1-31-59L (NO INTERFERENCE)

BASED ON ROTATING AREA

HUB 1 CYLINDRICAL HUB $C_D = .65$

HUB 2 TWO BLADED TEETERING HUB $C_D = .63$

HUB 3 HILLER SERVO ROTOR HUB $C_D = .70$

HUB 4 H-19 HUB $C_D = .72$

HUB 5 HUP-2 HUB $C_D = .55$

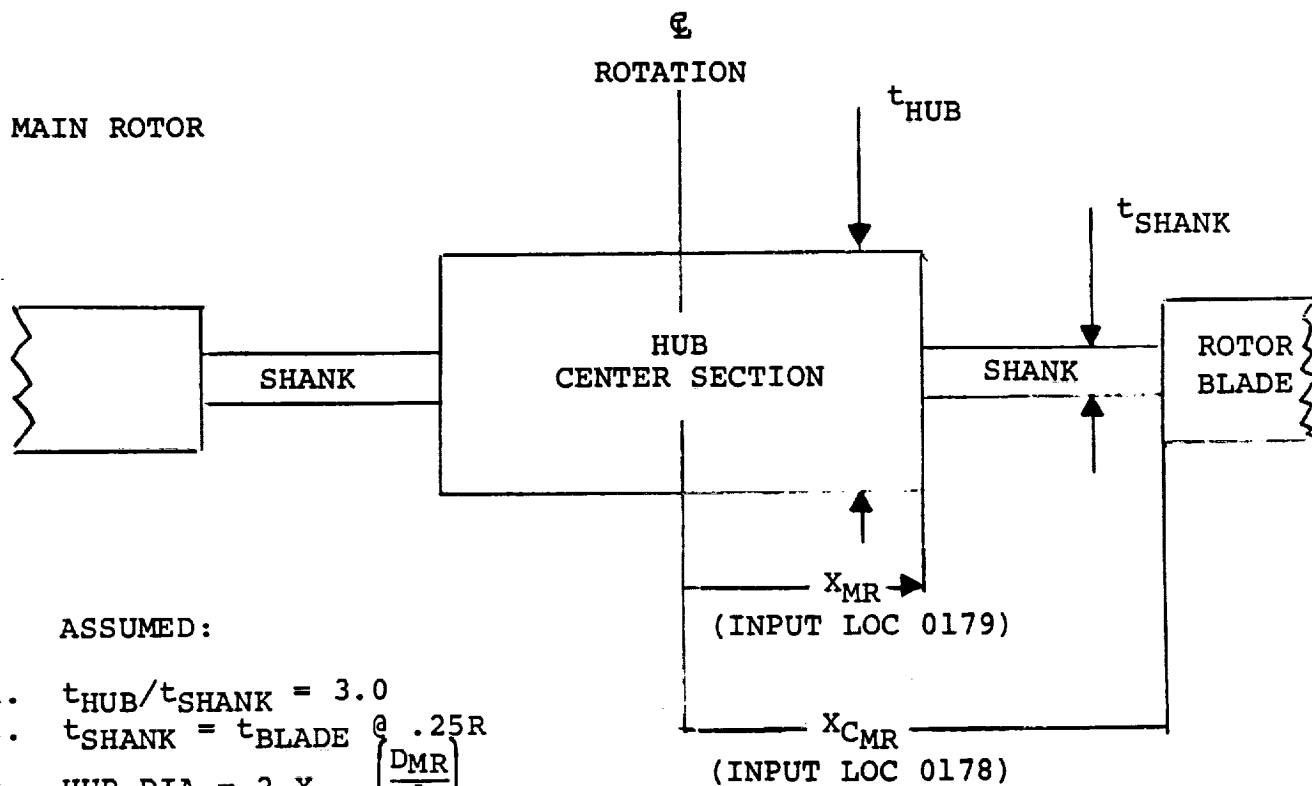
NOTE: HUB CENTERSECTION COEFFICIENTS ARE GENERALLY LOWER THAN MIGHT BE EXPECTED SINCE CENTERSECTION IS USUALLY MORE TYPICAL OF 3 DIMENSIONAL RATHER THAN 2 DIMENSIONAL FLOW

SHAPE

C_D @ SUBCRITICAL REYNOLDS NO.

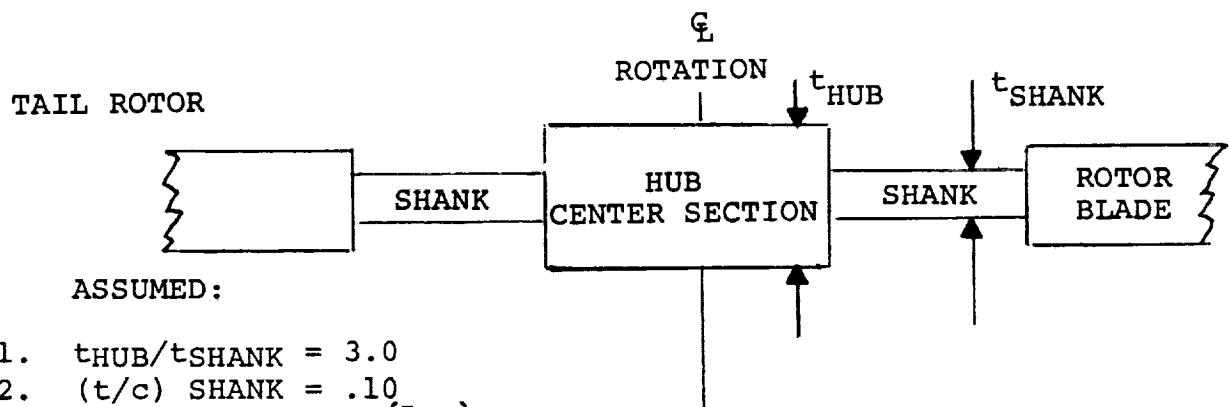
	3-DIMENSIONAL TYPICAL OF CENTERSECTION	2-DIMENSIONAL TYPICAL OF SHANKS
○	.47	1.17
◇	.80	1.55
□	1.05	2.05
	1.17	1.98

Figure 4-24. Typical Hub and Shank Drag Coefficients
(Part 2 of 2).



ASSUMED:

1. $t_{HUB}/t_{SHANK} = 3.0$
2. $t_{SHANK} = t_{BLADE} @ .25R$
3. $HUB\ DIA = 2 X_{MR} \left(\frac{DMR}{2} \right)$
4. $HUB\ FRONTAL\ AREA = HUB\ DIA \times t_{HUB}$
5. $SHANK\ FRONTAL\ AREA = (X_{CMR} - X_{MR}) \left(\frac{DMR}{2} \right) t_{SHANK}$
6. $K_{INT} = 1.00$



ASSUMED:

1. $t_{HUB}/t_{SHANK} = 3.0$
2. $(t/c)\ SHANK = .10$
3. $HUB\ DIA = 2 X_{TR} \left(\frac{D_{TR}}{2} \right)$
4. $HUB\ FRONTAL\ AREA = HUB\ DIA \times t_{HUB}$
5. $SHANK\ FRONTAL\ AREA = (X_{CTR} - X_{TR}) \left(\frac{D_{TR}}{2} \right) t_{SHANK}$
6. $K_{INT} = 1.00$

Figure 4-25. Rotor Hub/Shank Geometry Used in Program for Hub Drag Calculations.

TABLE 4-4. HUB AND SHANK DRAG COEFFICIENTS CORRECTION SUMMARY

MAIN ROTOR

$$C_{DCSMR} = C_{DCSMR} \left[\frac{(t_{HUB}/t_{SHANK})_{MR}}{3.0} \right] \left[\frac{(t/C)_{SHANK}}{(t/C)_{.25R}} \right] \left[\frac{X_{MR}}{X_{MR}} \right] \left[\frac{K_{INT}}{1.00} \right]^*$$

OBT FROM FIG 4-25

INPUT LOC 0180 INPUT LOC 0179

$$C_{DSHMR} = C_{DSHMR} \left[\frac{(t/C)_{SHANK}}{(t/C)_{.25R}} \right] \left[\frac{(X_{CMR} - X_{MR})}{(X_{CMR} - X_{MR})} \right] \left[\frac{K_{INT}}{1.00} \right]^*$$

INPUT LOC 0180 INPUT LOC 0178 INPUT LOC 0179

TAIL ROTOR

$$C_{DCSTR} = C_{DCSTR} \left[\frac{(t_{HUB}/t_{SHANK})_{TR}}{3.0} \right] \left[\frac{(t/C)_{SHANK}}{.10} \right] \left[\frac{K_{INT}}{1.00} \right]^*$$

OBT FROM FIG. 4-25

$$C_{DSHTR} = C_{DSHTR} \left[\frac{(t/C)_{SHANK}}{.10} \right] \left[\frac{K_{INT}}{1.00} \right]^*$$

*NUMERATORS SHOW ACTUAL HUB/SHANK CHARACTERISTICS DESIRED.
DENOMINATORS SHOW FIXED CHARACTERISTICS.

where,

W_{G0} = Initial "guess" at iterated helicopter gross weight (input LOC 0023). Note, this is also the value of gross weight at which the input value of (GW/Fe) is obtained. (See sketch below)

W_G = "Iterated" or final design gross weight of the aircraft.

(GW/Fe) = "Drag Loading" input at a given gross weight (in this case W_{G0}).

K_{FED} = Exponent defining the slope of a typical logarithmic drag trend.

The following sketch should serve to illustrate these facts more clearly.

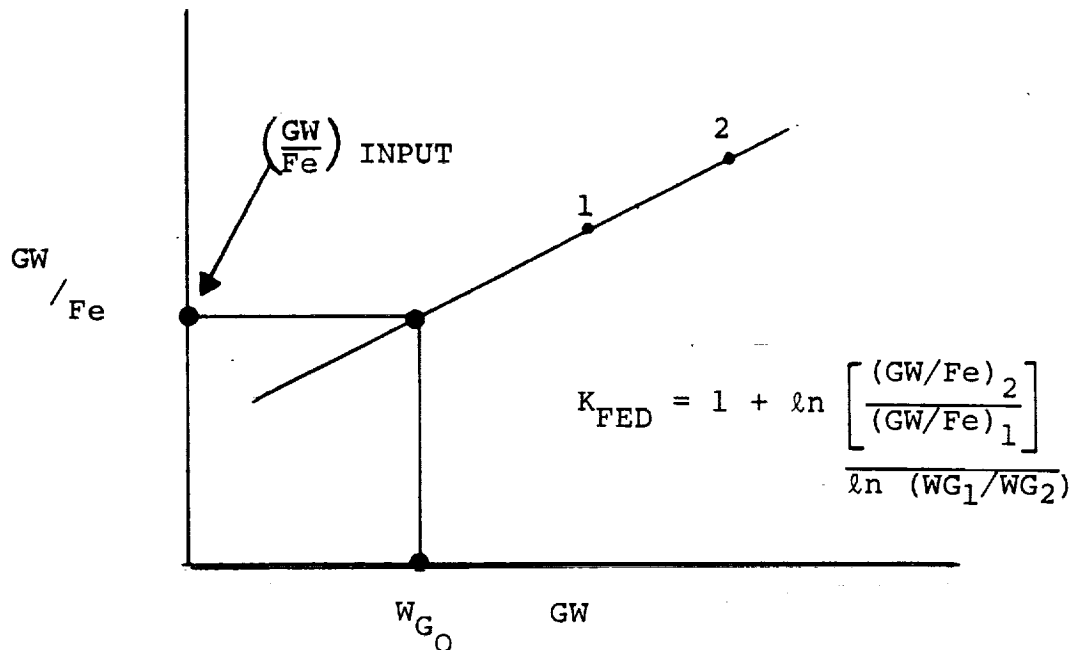
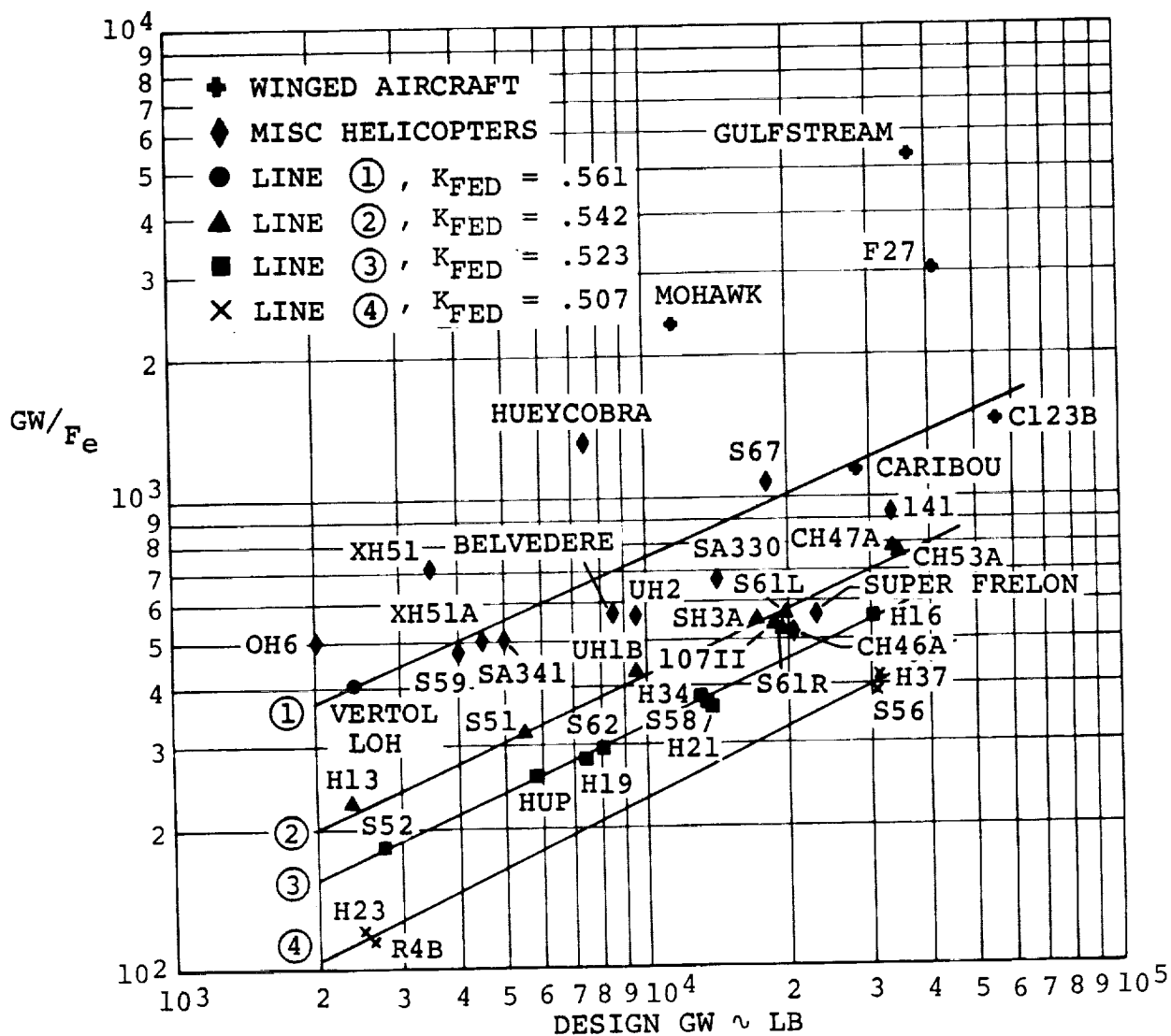


Figure 4-26 illustrates typical parasite drag area trend for various helicopters and fixed-wing aircraft.

The drag routine may be used in many different ways. The four most common applications are:

1. Drag Build up for a New Aircraft Design - This is best illustrated by first referring to the complete drag breakdowns of the hypothetical helicopters shown in Tables 4-5 and 4-7. The input C_D for each component (C_{DWi} , $DHTi$,



1. Advanced drag cleanup (e.g. faired hubs, low drag or retractable landing gear etc.).
2. Current standard of landing gear, hub design, skin finish etc.
3. Unfaired landing gear, hubs, protuberances, poor body shape, etc.
4. Exceptionally dirty configuration due to such things as open construction, exceptionally dirty engine installation, landing gear, etc.

NOTE: The drag area for the winged aircraft excludes the C_{DA} of the wings, to be compatible with the helicopters.

Figure 4-26. Typical Parasite Drag Trends.

etc.) may be used to represent the reference C_f at $Re = 10^7$ and at the mean flight Mach number. Drag increases above the drag of a flat plate such as three dimensional effects, interference, roughness, and excrescences may be accounted for by the multiplying factors (K_W , K_{HT} , etc.). Miscellaneous drag increments can be summed and input as ΔFe . Examples of these increments are cooling momentum, trim, and airconditioning. The K factor for wings and tails should include a factor for relating the wetted area of the surface to the planform area. An example of the program inputs for the hypothetical helicopters of Tables 4-5 and 4-7 are shown in Tables 4-6 and 4-8.

2. Study of the Sensitivity of Aircraft Size with Respect to the Component Drag or the Total Drag about a Certain Drag Level - Let the total drag of each component be contained in the drag coefficient of each component, C_{Dwi} , C_{DHTi} , C_{DVTi} , etc. The change in drag of each component will then be determined by the values assigned to the component multiplying factor, K_W , K_{HT} , K_{VT} , etc. The fuselage drag change, however, will have to be represented by an incremental value of Δfe .
3. Use of Component Drag Data from Wind Tunnel Test - Let the drag of each component (including interference) be contained in the component drag coefficient, C_{Dwi} , C_{DHTi} , C_{DVTi} , etc. The skin friction drag must first be corrected to $Re = 10^7$. The drag increase due to items found only on the full scale airplane would then be represented by the factors and increments. Increases due to excrescences and roughness are represented by the factors, K_W , K_{HT} , K_{VT} , etc. Increments such as inlets, cooling, trim, and after-body drag, can be summed and represented by Δfe .
4. Simplified Drag Model for Parametric Studies - The program is often used to study the influence of variations of parameters, such as disc loading, solidity, etc. on the size of a helicopter. During these studies, it is often not desirable to go into the design depth required in the three applications above. Use of the drag trends ($DRGIND = 2$) is therefore dictated by this requirement.

Figure 4-27 is a flow chart of this subroutine.

TABLE 4-5. DRAG BREAKDOWN FOR HYPOTHETICAL SINGLE ROTOR COMPOUND HELICOPTER $R_e/FT = 1.56 \times 10^6$ ($V = 189$ KT)

COMPONENT	WETTED AREA	C_f	INCREMENT		f_e FT ²
			%	Δf_e	
FUSELAGE	1949.	0.00191		3.72	6.60
3-Dimensional Effects			12.3	0.46	
Excrescences			7.0	0.29	
Canopy				0.20	
Afterbody				1.93	
WING	345.	0.0030		1.04	1.59
3-Dimensional Effects			29.3	0.31	
Excrescences			2.0	0.03	
Flaps, Slats, Ailerons, Spoilers			-	-	
Body Interference			15.6	0.21	
MAIN ROTOR PYLON	286.				3.19
Basic f_e (including interference and 3-D effects) C_{D_0} (Based on frontal area) = 0.10	Frontal Area = 26.6 Ft ²			2.66	
Excrescences				20.0 0.53	
HORIZONTAL TAIL	246.	0.00305		0.75	1.60
3-Dimensional Effects			43.4	0.33	
Excrescences			7.0	0.08	
Interference			40.7	0.44	
VERTICAL TAIL	210.	0.00270		0.57	1.03
3-Dimensional Effects			22.3	0.13	
Excrescences			7.0	0.05	
Interference			40.7	0.28	
PRIMARY ENGINE NACELLES	188.	0.00278		0.52	1.72
3-Dimensional Effects			40.4	0.21	
Excrescences			25.0	0.18	
Interference			75.0	0.55	
Inlets				0.26	
ROTOR HUBS (TOTAL)					14.82
Main Rotor Hub (center section)				6.22	
Main Rotor Hub (shanks)				5.85	
Tail Rotor Hub (center section)				0.48	
Tail Rotor Hub (shanks)				0.67	
Total Interference (main & tail rotor)				1.60	
MISC					1.34
Roughness (50% of $C_f A_{WET}$)				0.54	
Cooling				0.50	
Trim				-	
Air Conditioning				0.30	
TOTALS ft ²	3224				31.89
NOTES: (1) Basic $f_e = (C_f A_{WET}) + (3-D \text{ Effects } \Delta f_e)$					
(2) Excrescences and interference are % of basic f_e					

TABLE 4-6. SUMMARY OF AERODYNAMICS INPUT FOR COMPOUND HELICOPTER OF TABLE 4-5	
GENERAL	$C_{D_{wi}} = C_{D_{HT}} = C_{D_{VT}} = C_{DN} = C_F = 10^7 (R_e/4)_i = 1.56 \times 10^6$
COMPONENTS	FUSELAGE WETTED AREA
FUSELAGE	$K_F = (1+0.123) \uparrow \text{3-D effects} \text{ excr } \uparrow \text{roughness} = 1.25 \Delta F_e = \left(\frac{0.00287}{0.00191} \right) \uparrow \text{Body } C_F \uparrow \text{Afterbody} \uparrow \text{Canopy} \uparrow \text{Nacelle inlets} \uparrow \text{Cooling} \uparrow \text{Air Condit.}$ $(1.93 + 0.20 + 0.26 + 0.50 + 0.30) = 4.80$
WING	$K_W = (1.80) \uparrow \left[(1 + 0.293) \uparrow \text{3-D effects} \uparrow \text{excr } \uparrow \text{roughness} \right] (1 + 0.02 + 0.156) + 0.05 \uparrow \text{interf } \uparrow \text{roughness} = 2.83$ (A_{wet}/S_w)
MAIN PYLON	$K_{FP} = 1 + 0.20 + 0.05 \uparrow \text{excr } \uparrow \text{roughness} = 1.25 C_{D_{FP}} = 0.10$
HOR VERTICAL TAIL	$K_{HT} = 2.02 \uparrow \left[(1 + 0.434) \uparrow \text{3-D effects} \uparrow \text{excr } \uparrow \text{interf } \uparrow \text{roughness} \right] + 0.05 \uparrow \text{roughness} = 4.38$ $(A_{wet}/S_{HT}) \uparrow \text{3-D effects} \uparrow \text{excr } \uparrow \text{interf } \uparrow \text{roughness}$ (A_{wet}/S_{VT}) $K_{VT} = 2.06 \uparrow \left[(1 + 0.223) \uparrow \text{3-D effects} \uparrow \text{excr } \uparrow \text{interf } \uparrow \text{roughness} \right] + 0.05 \uparrow \text{roughness} = 3.83$
PRIMARY ENGINE NACELLES	$K_N = (1 + 0.404) \uparrow \text{3-D effects} \uparrow \text{excr } \uparrow \text{interf } \uparrow \text{roughness} + 0.05 \uparrow \text{roughness} = 2.86$
ROTOR HUBS	$C_{DC_{SMR}} = 0.75$ $C_{D_{SHMR}} = 1.4$ $K_{HPIM} = 1.3$ $C_{DCSTR} = 0.75$ $C_{D_{SHTR}} = 1.4$ $K_{HPSM} = 0.3$
MISC	Cooling, Airconditioning, and nacelle inlets included in fuselage ΔF_e

TABLE 4-7. TYPICAL DRAG SUMMARY
DRAG BREAKDOWN FOR HYPOTHETICAL TANDEM ROTOR
WINGED HELICOPTER $R_e/ft = 1.49 \times 10^6$ ($V = 181$ KT)

COMPONENT	WETTED AREA	C_f	INCREMENT		f_e FT ²
			%	f_e	
FUSELAGE	1640.	0.00208		3.41	6.52
3-Dimensional Effects			12.2	0.42	
Excrescences			7.0	0.27	
Canopy				0.20	
Afterbody				2.22	
WING	389.	0.0030		1.16	1.77
3-Dimensional Effects			29.3	0.34	
Excrescences			2.0	0.03	
Flaps, Slats, Ailerons, Spoilers			-	-	
Body Interference			15.2	0.24	
FORWARD PYLON	41.5				1.69
Basic f_e (including interference and 3-D effects) C_{D_0} (Based on Frontal Area) = 0.15	(Frontal Area = 9.5 Ft ²)			1.41	
Excrescences			20.0	0.28	
AFT PYLON	421.	0.00252		1.06	2.38
3-Dimensional Effects			46.4	0.49	
Excrescences			20.0	0.31	
Interference			33.5	0.52	
PRIMARY ENGINE NACELLE	188.	0.00278		0.52	1.72
3-Dimensional Effects			40.4	0.21	
Excrescences			25.0	0.18	
Interference			75.0	0.55	
Inlets				0.26	
ROTOR HUBS (TOTAL)					19.48
Main rotor hub (center section, total)				10.4	
Main rotor hub (shanks, total)				7.78	
Hub/Shank Interference (total)				1.30	
MISC					1.25
Roughness (5.0% of $C_{f,WET}$)				0.45	
Cooling				0.50	
Trim				-	
Air Conditioning				0.30	
TOTALS ft ²	2679.5				34.81
NOTES: (1) Basic $f_e = (C_{f,WET}) + (3-D \text{ Effects } \Delta f_e)$ (2) Excrescences and interference are % of basic f_e					

TABLE 4-8. SUMMARY OF AERODYNAMICS INPUT FOR HELICOPTER OF TABLE 4-7

GENERAL			
$C_{D_{wi}} = C_{D_{AP}} = C_{DN} = C_F = 0.00287 @ R_e = 10^7 \quad (Re/\rho)_1 = 1.49 \times 10^6$			
COMPONENTS	$S_F = 1640$	FUSELAGE WETTED AREA	cooling air condit
FUSELAGE	$K_F = (1+0.122)$ 3-D effects	$(1+0.07)$ excr $+0.05$ roughness $\Delta F_e = \left(\frac{0.00287}{0.00208} \right)$ Body C_F Afterbody Canopy Nacelle inlets	$(2.22+0.20+0.26+0.50+0.30)=4.81$ ✓
WING	$K_w = (1.80)$ $\left(\frac{A_{wet}}{S_w} \right)$	$[(1 + 0.34) \uparrow$ 3-D effects $(1 + 0.02 + 0.152) \uparrow$ excr interf $+ 0.05] \uparrow$ roughness	$= 2.92$
FORWARD PYLON	$K_{FP} = 1 + 0.20 + 0.05 + 1.25$ excr roughness	$C_{DFP} = 0.15$	
AFT PYLON	$K_{AP} = (0.455)$ $\left(\frac{S_{PAP}}{A_{wet}} \right)$	$[(1 + 0.464) \uparrow$ 3-D effects $(1 + 0.20 + 0.335) \uparrow$ excr interf $+ 0.05] \uparrow$ roughness	$= 1.045$
PRIMARY ENGINES NACELLES	$K_N = (1 + 0.404)$ 3-D effects	$[1 + 0.25 + 0.75] \uparrow$ excr interf $+ 0.05 \uparrow$ roughness	$= 2.86$
ROTOR HUBS	$C_{D_{CSMR}} = 0.75$ $C_{D_{SHMR}} = 1.4$	$K_{HPIM} = 1.3$	
MISC	Cooling, air conditioning and nacelle inlets included in fuselage ΔF_e		

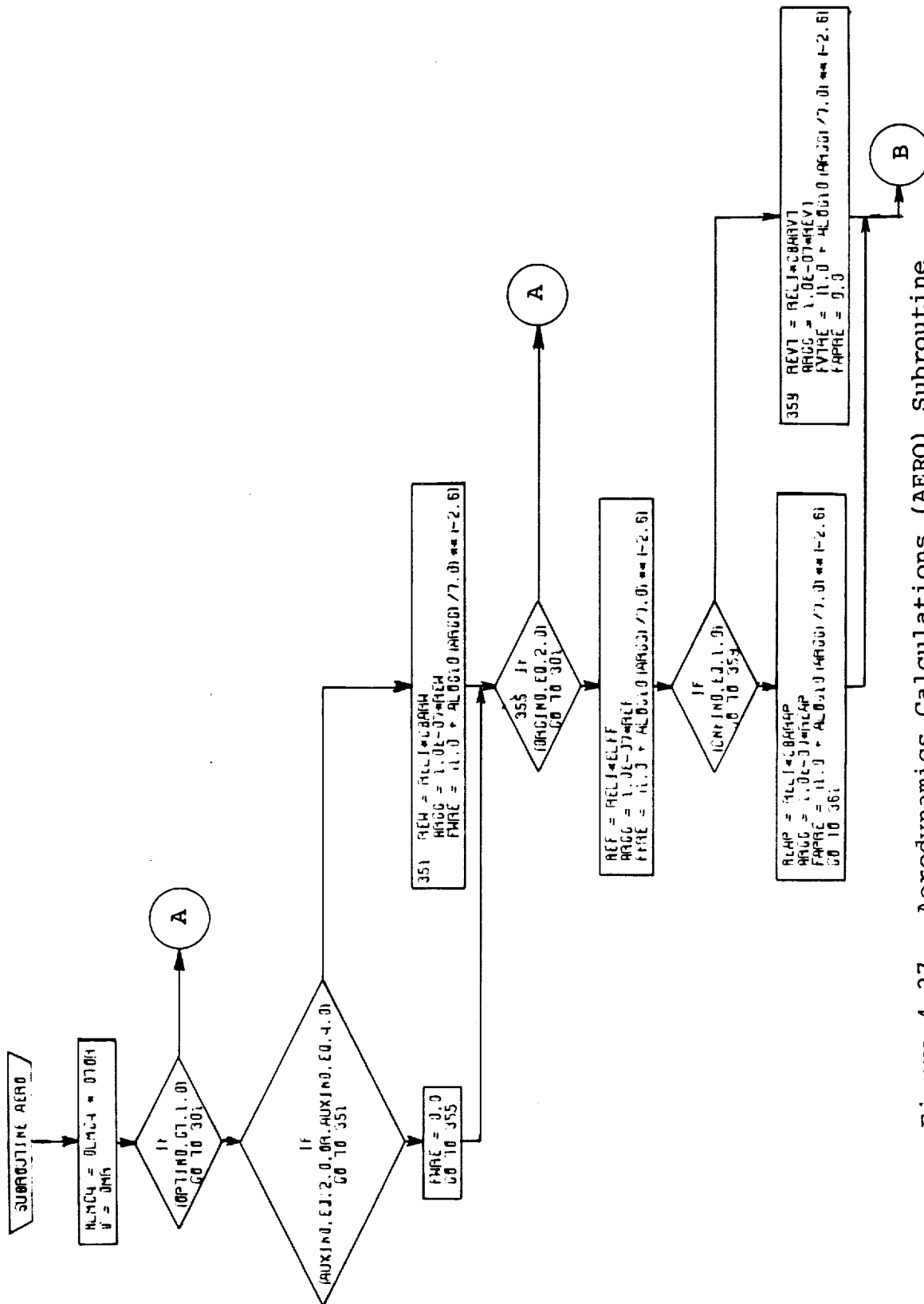


Figure 4-27. Aerodynamics Calculations (AERO) Subroutine
Flow Chart (Part 1 of 4).

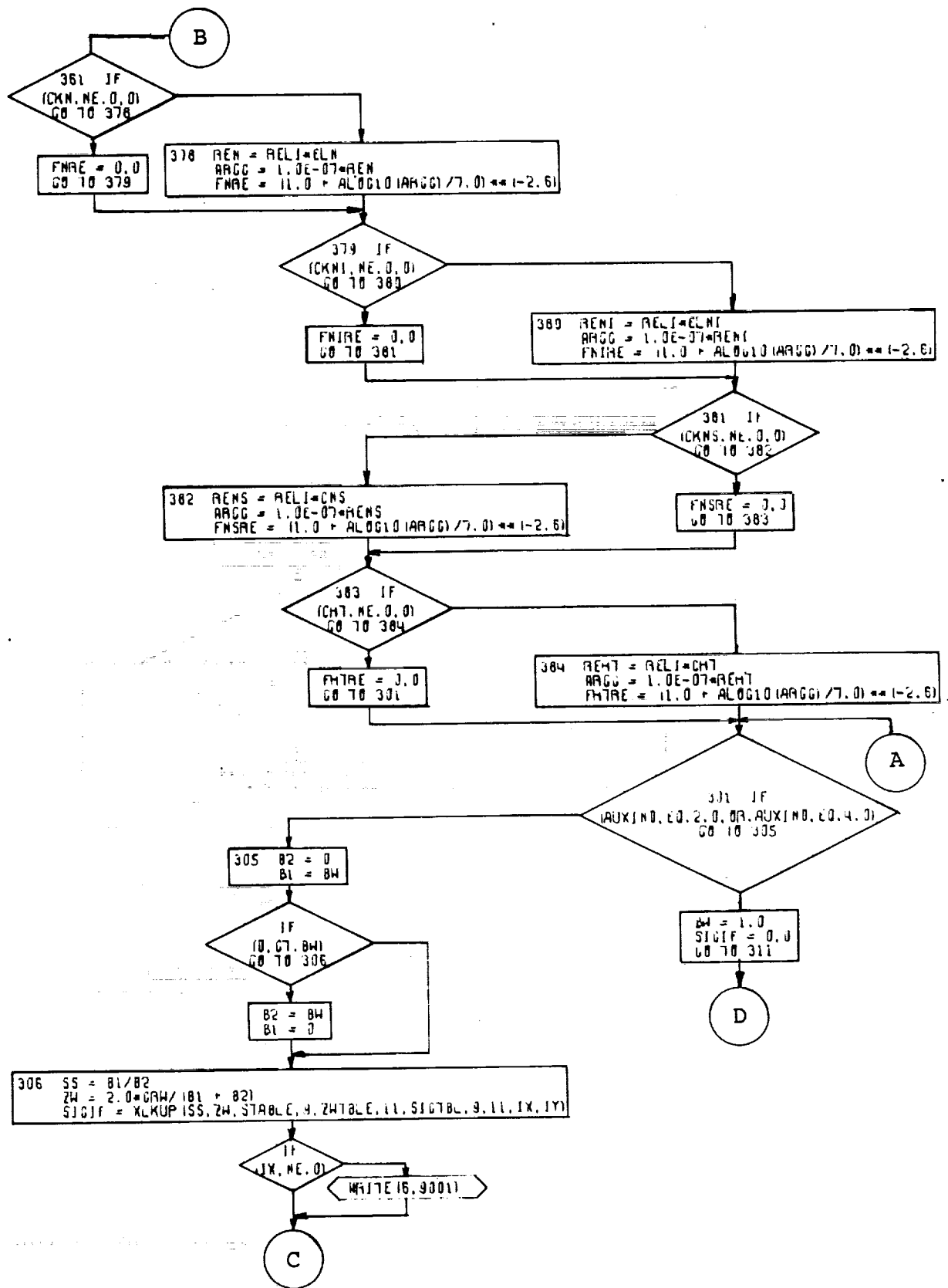


Figure 4-27. Aerodynamics Calculations (AERO) Subroutine Flow Chart (Part 2 of 4).

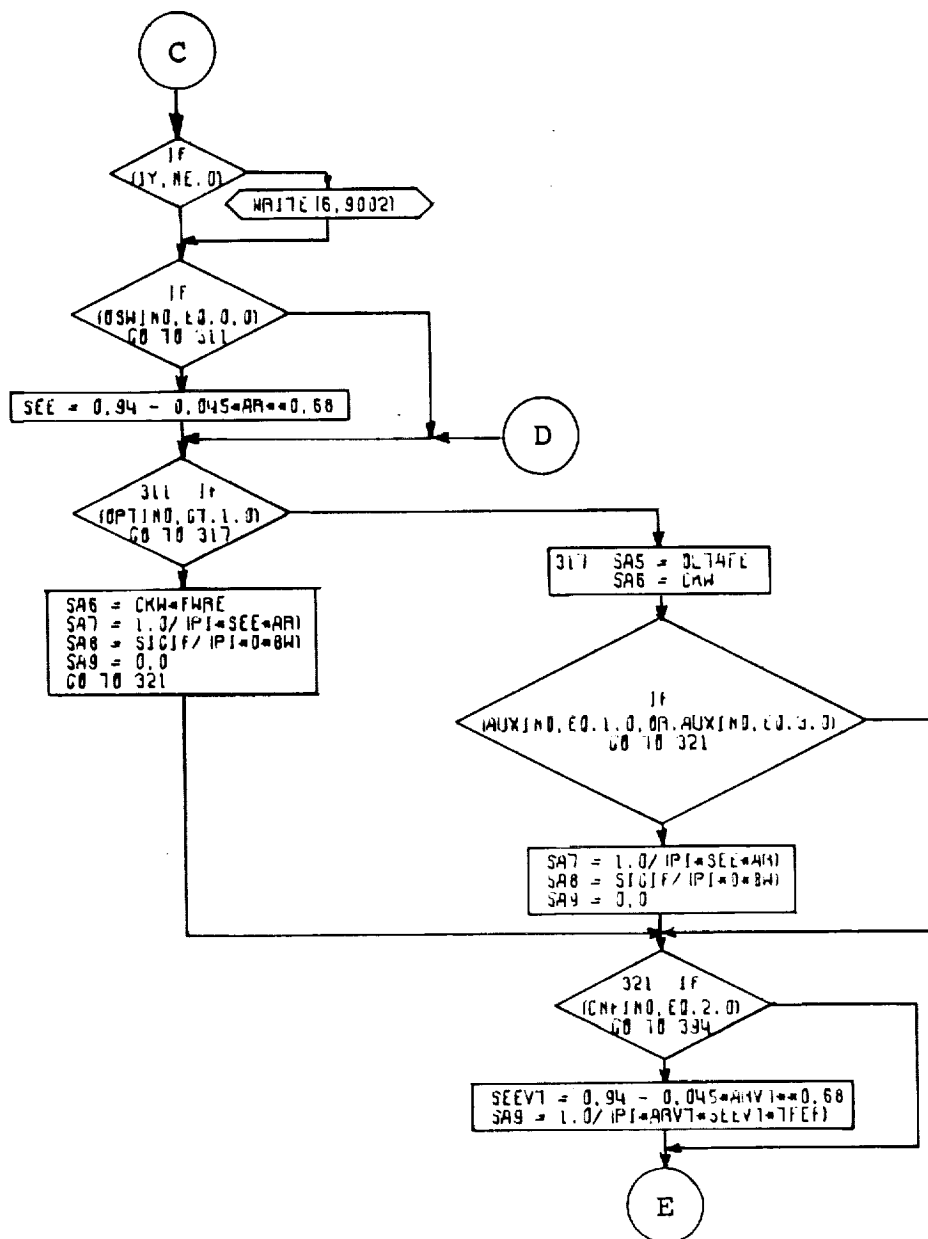


Figure 4-27. Aerodynamics Calculations (AERO) Subroutine Flow Chart (Part 3 of 4).

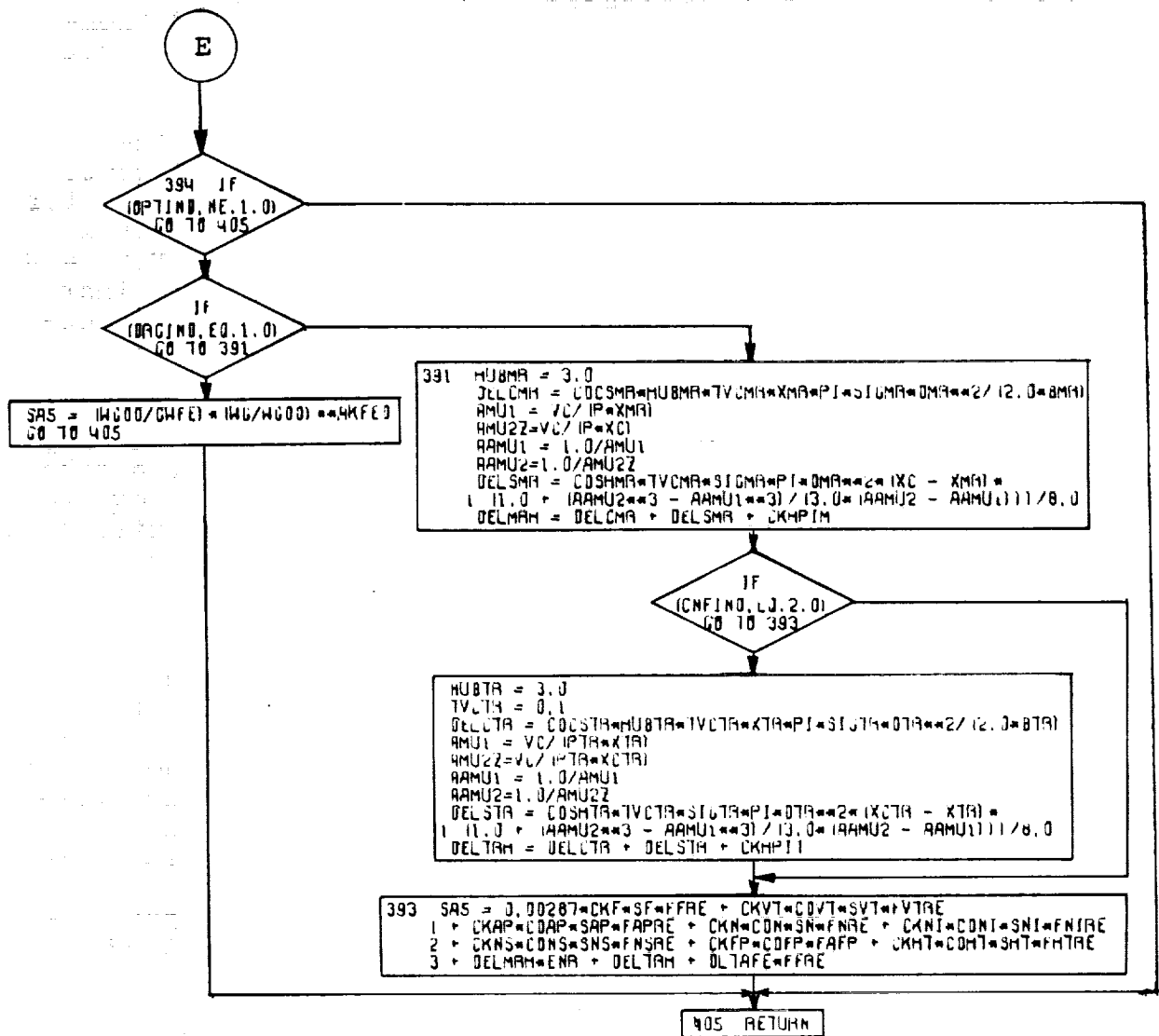


Figure 4-27. Aerodynamics Calculations (AERO) Subroutine Flow Chart (Part 4 of 4).

4.10 ENGINE SIZING SUBROUTINE

The engine cycle performance data included in the engine library consists of detailed performance maps of power (or thrust), fuel flow, N_I , and N_{II} . The data, as shown in Table 4-1, is in normalized, referred format. In particular, horsepower is normalized with respect to the value of power at the maximum static rating at sea level, standard day conditions. Thrust is similarly normalized to the maximum static thrust at sea level for standard day.

The engine sizing subroutine calculates the value of the scaling factors; namely, the maximum static thrust or power (S.L., std.). If so desired, the user may study a helicopter with fixed rather than "rubberized" primary engines. This is accomplished by means of the indicator FIXIND. If FIXIND = 0, the user inputs the maximum (installed) power of the primary engines. If FIXIND = 1, the engine sizing subroutine calculates the installed power.

A variety of different criteria are often applied to determine engine size requirements. These criteria, differing as they do, can generally be related by a single factor. For a take-off condition this factor is the value of equivalent required thrust-to-weight ratio. Similarly, engine sizing requirements for forward flight can be related to a set of cruise conditions; namely, cruise altitude and true airspeed.

Engine sizing requirements for helicopters are generally set by takeoff conditions, and less frequently by forward flight conditions. The program, therefore, permits the user two options of calculation. The first option (ESCIND = 1) will calculate engine size for takeoff conditions only; the second option (ESCIND = 2) will calculate engine size for both takeoff and forward flight conditions, compare the two, and pick the more critical condition. The engines are sized for takeoff to provide a required (input) equivalent thrust-to-weight ratio with a specified (input) number of engines inoperative, at a specified fraction (SHP_E/SHP^*) of maximum installed power.

Cruise conditions are specified by means of altitude, ambient temperature, true airspeed; and, in the case of a compound helicopter, the propulsive thrust split (T_{AUX}/T_{TOT})_C between the auxiliary propulsor and the main rotor. In addition, the user may select the power setting to be used: maximum, military, or normal.

In the case of a configuration having auxiliary independent cruise engines (APIND = 2), these engines may either be fixed or sized independently of the primary engines. For example, it would be possible to study a configuration with fixed size

primary engines (FIXIND = 0) while sizing the auxiliary engines to meet cruise requirements. NOTE: in a case like this, input locations 0234-0241 must be filled out to allow sizing for the auxiliary engines, even though the primary engines are fixed in size.

In addition to sizing the primary and auxiliary engines, this subroutine calculates the main rotor, tail rotor (in the case of a single rotor helicopter) and auxiliary propulsion drive system rating (in the case of a compound helicopter). The options available to the user for this purpose are:

- XMSNIND = 1 Main, tail and auxiliary drive system ratings specified as fraction of primary engine installed power (in the case of a compound helicopter with auxiliary independent propulsion, the auxiliary independent drive system rating is specified as a fraction of the auxiliary independent engine installed power)
- XMSNIND = 2 Main, tail, and auxiliary drive system ratings specified at fraction of power required to hover or cruise at design conditions (more critical of the two conditions is selected).
- XMSNIND = 3 Same as 2, except the most critical of the two design condition is compared to the drive system rating required at an alternate payload/gross weight hover at the design point conditions, the most critical of these three conditions is selected.

It should be noted that when FIXIND = 0 or FIXINDI = 0; i.e., fixed size engines, the drive system ratings calculated are equal to the product of the applicable multiplicative factors ($\text{SHP}_{\text{MRX}}/\text{SHP}^*_{\text{MR}}$, $\text{SHP}_{\text{TRX}}/\text{TRP}^*$, $\text{SHP}_{\text{ARX}}/\text{SHP}^*_{\text{AUX}}$) and the input fixed engine sizes. If FIXIND = 1.0 or FIXINDI = 1.0 and XMSNIND = 1.0, the drive system ratings calculated are equal to the product of the applicable multiplicative factors ($\text{SHP}_{\text{MRX}}/\text{SHP}^*_{\text{MR}}$, $\text{SHP}_{\text{TRX}}/\text{TRP}^*$, $\text{SHP}_{\text{ARX}}/\text{SHP}^*_{\text{AUX}}$) and the component (main tail, and auxiliary) power obtained from the proportional split (based on power required) of the total sea level standard maximum (installed) engine power.

The use of separate engine and transmission sizing options provides great flexibility in meeting conflicting engine/drive system requirements. For example, using XMSNIND = 2, it is possible to size a helicopter's primary engines to meet an engine inoperative in hover requirement, while only rating the drive system for the actual power required to hover at that design point, thus effecting a considerable saving in drive system weight. Or, using XMSNIND = 3, it is possible to rate

the drive system for the power required at an alternate gross weight/payload hover point, while still meeting the original engine out sizing criteria. Figure 4-28 contains a flow chart of this subroutine.

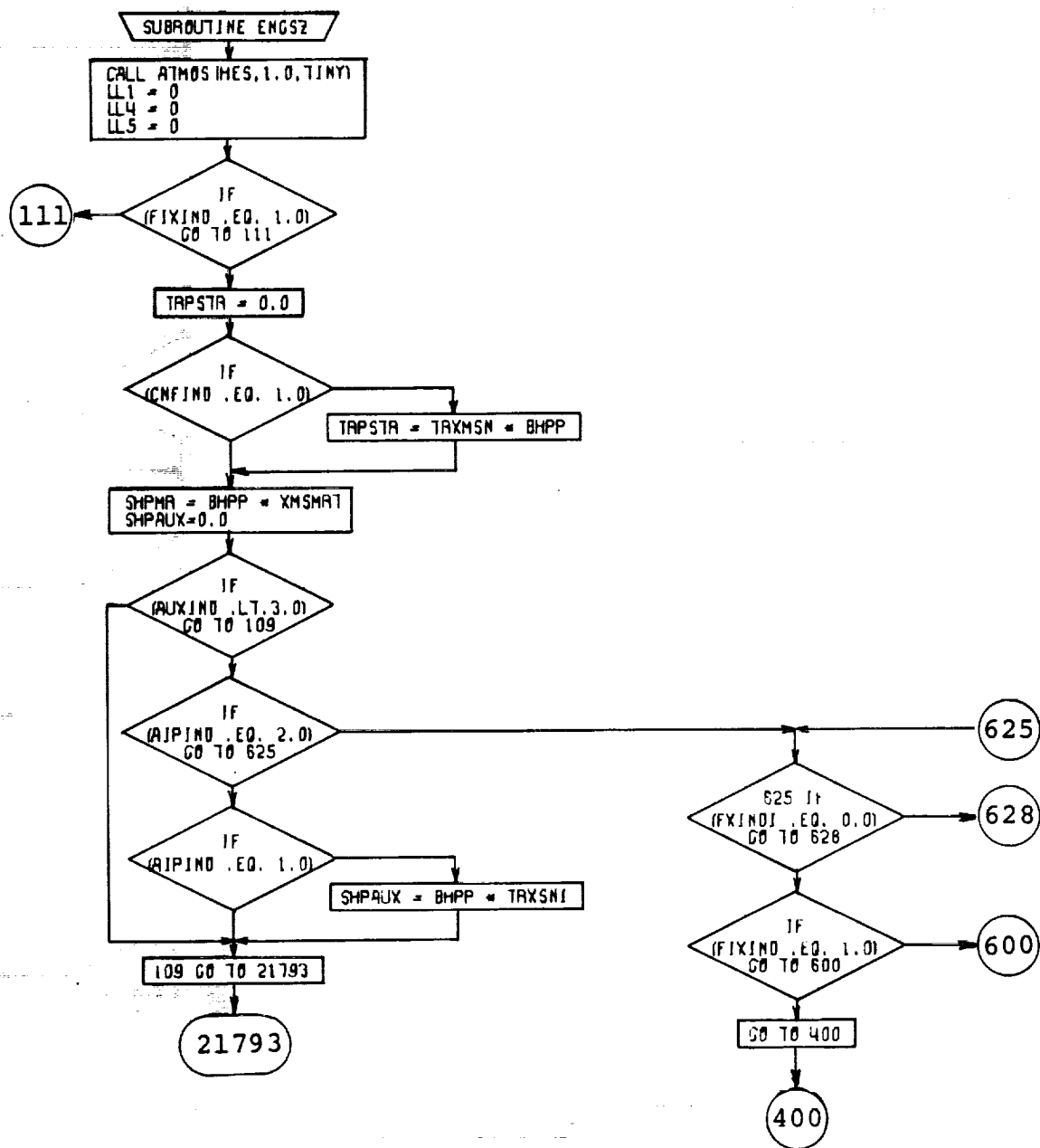


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 1 of 10).

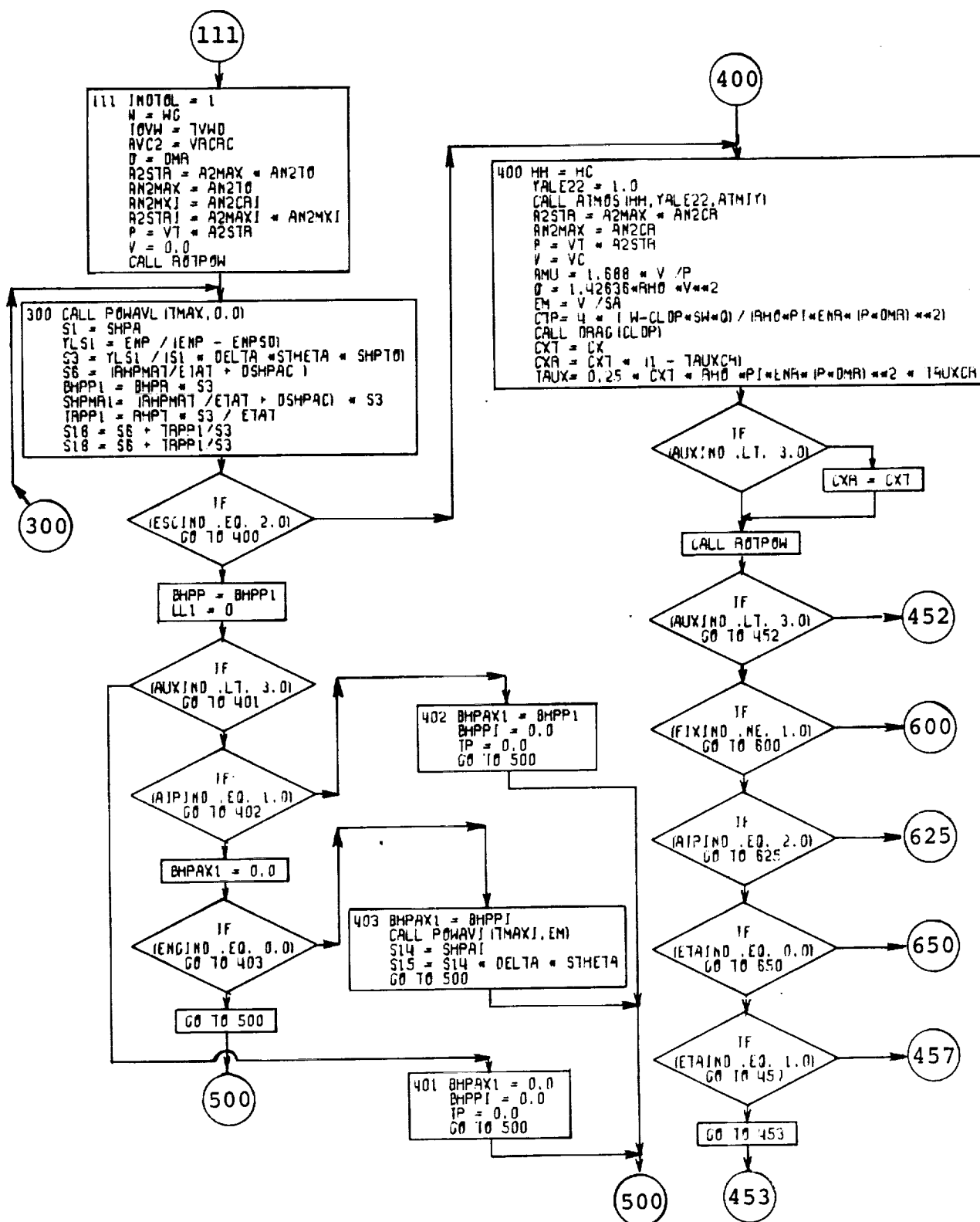


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 2 of 10).

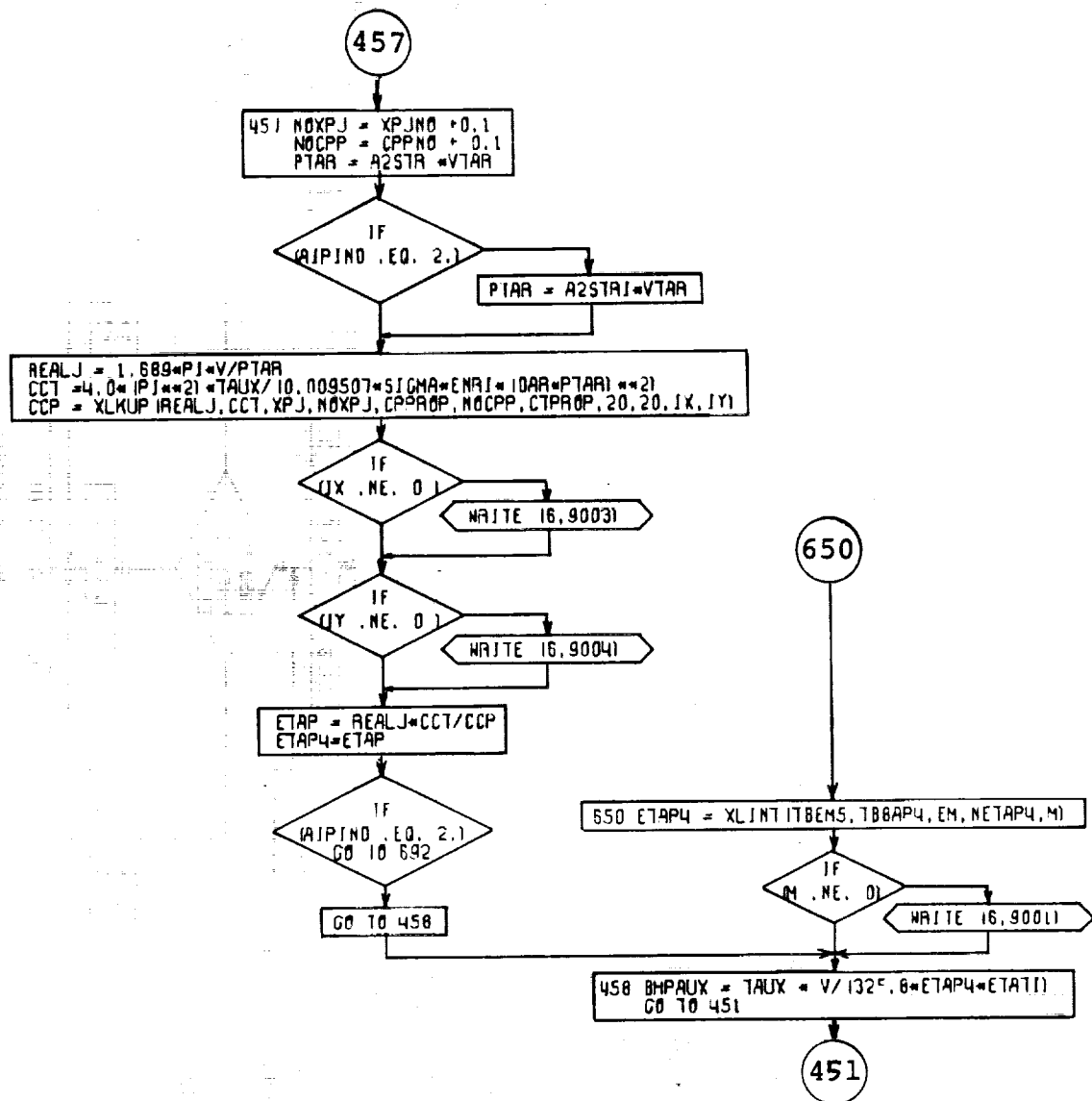


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 3 of 10).

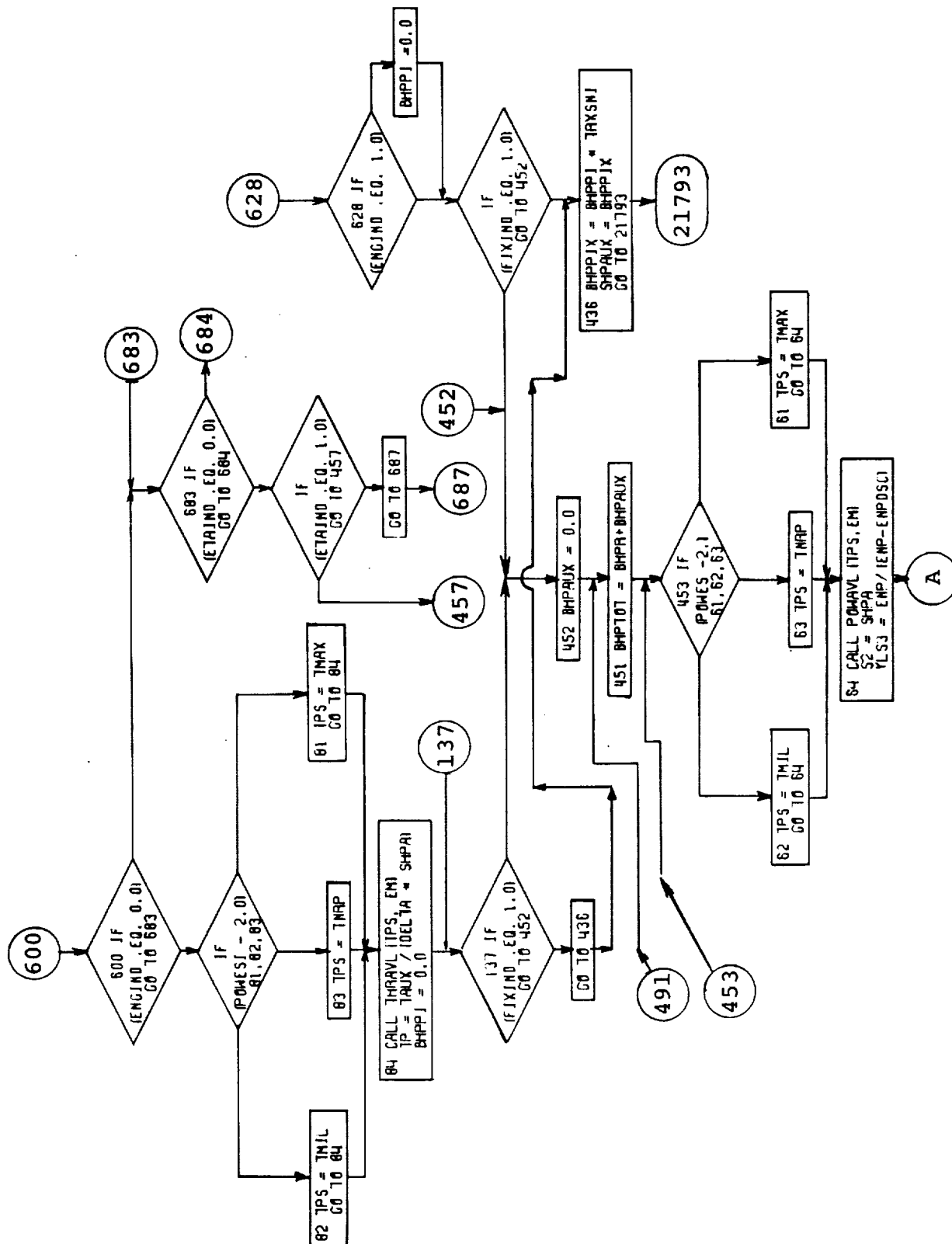


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 4 of 10).

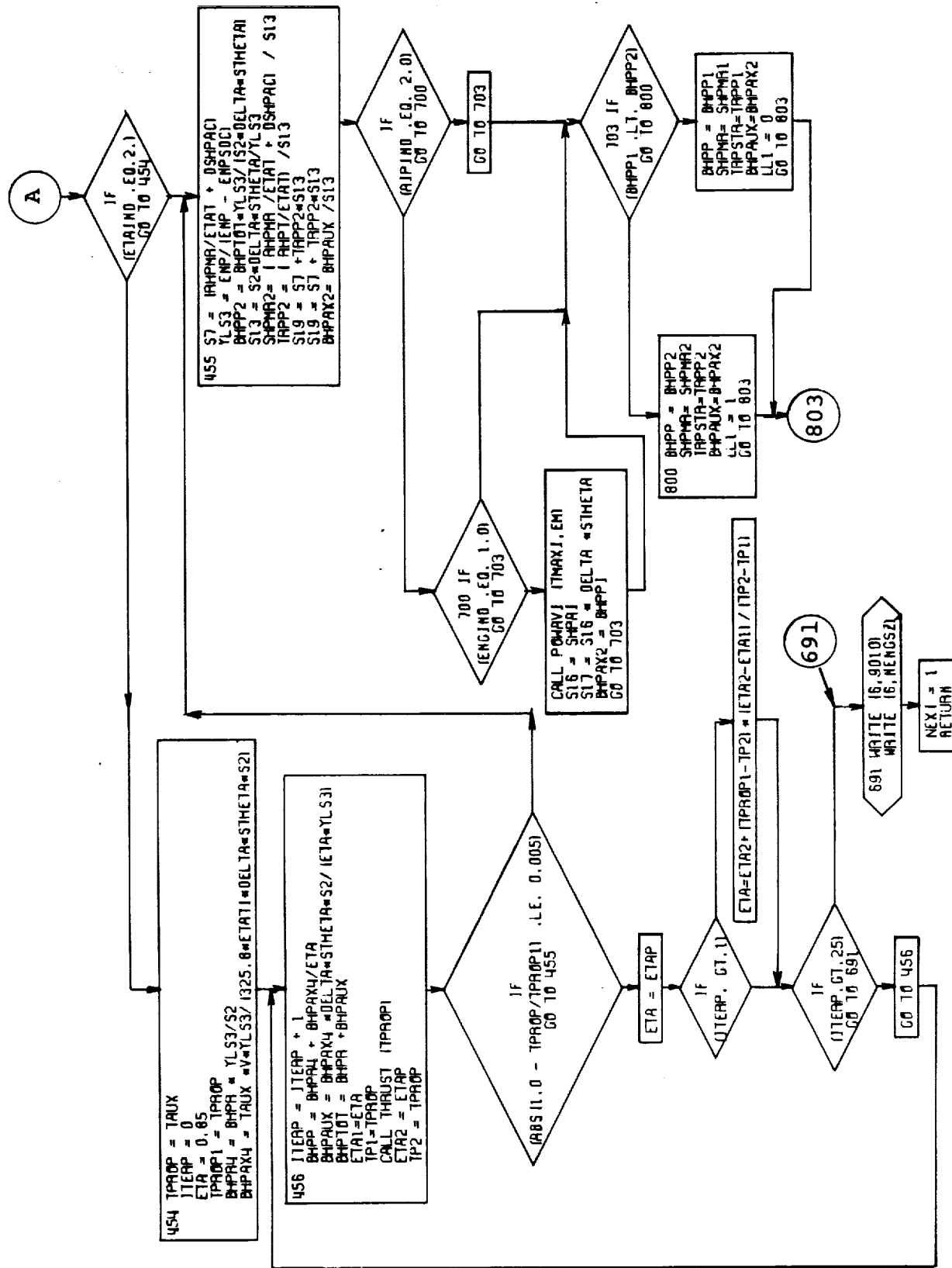


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 5 of 10).

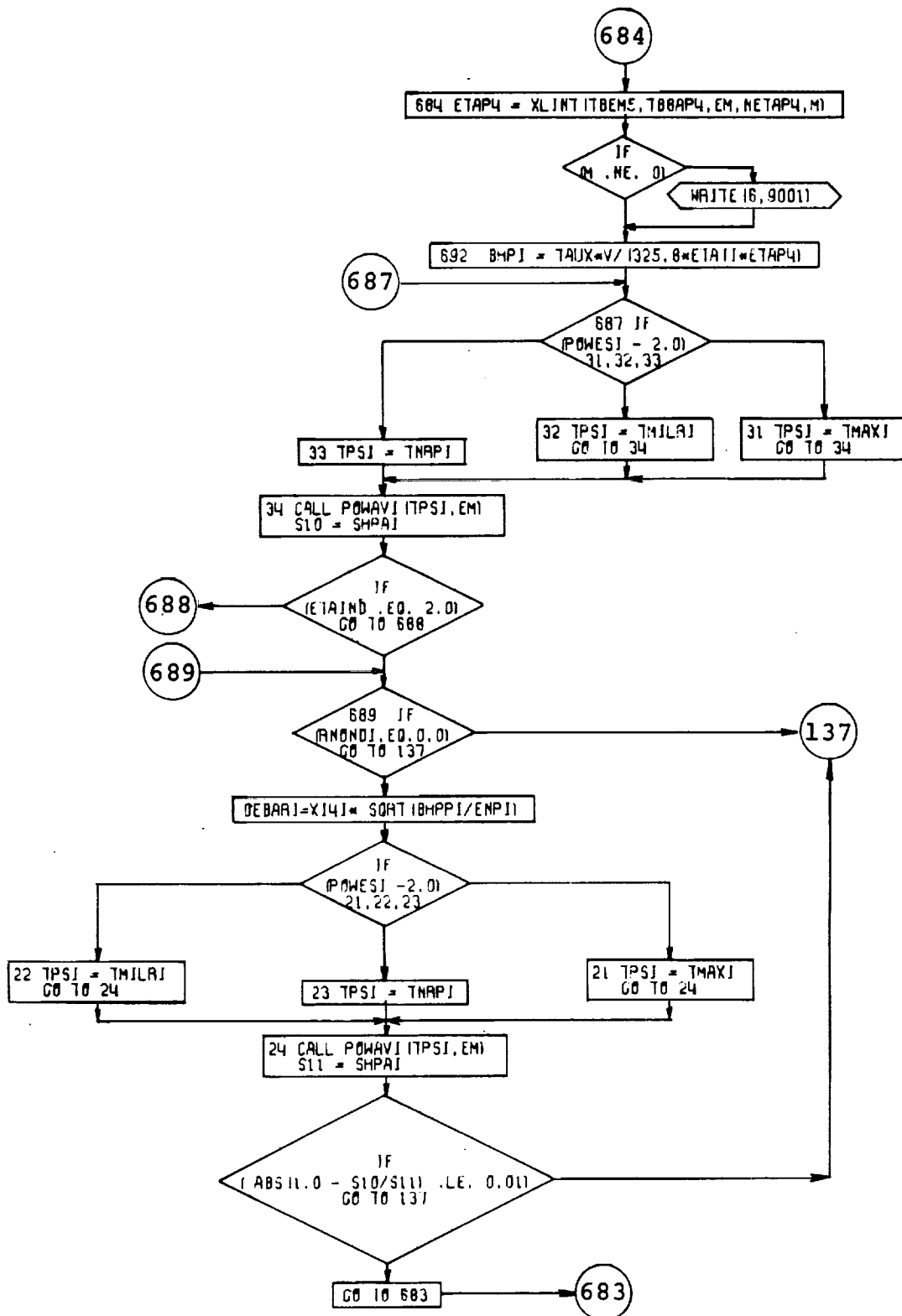


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 6 of 10).

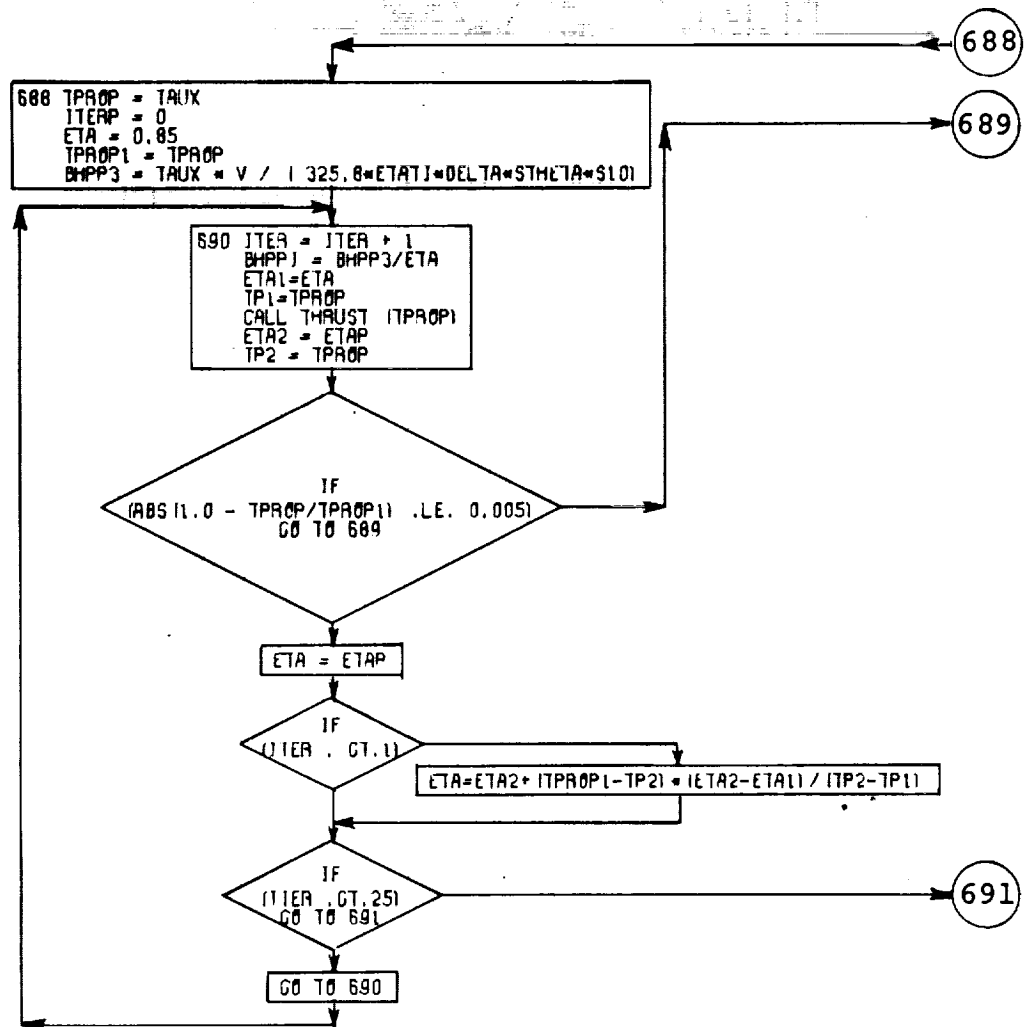


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 7 of 10).

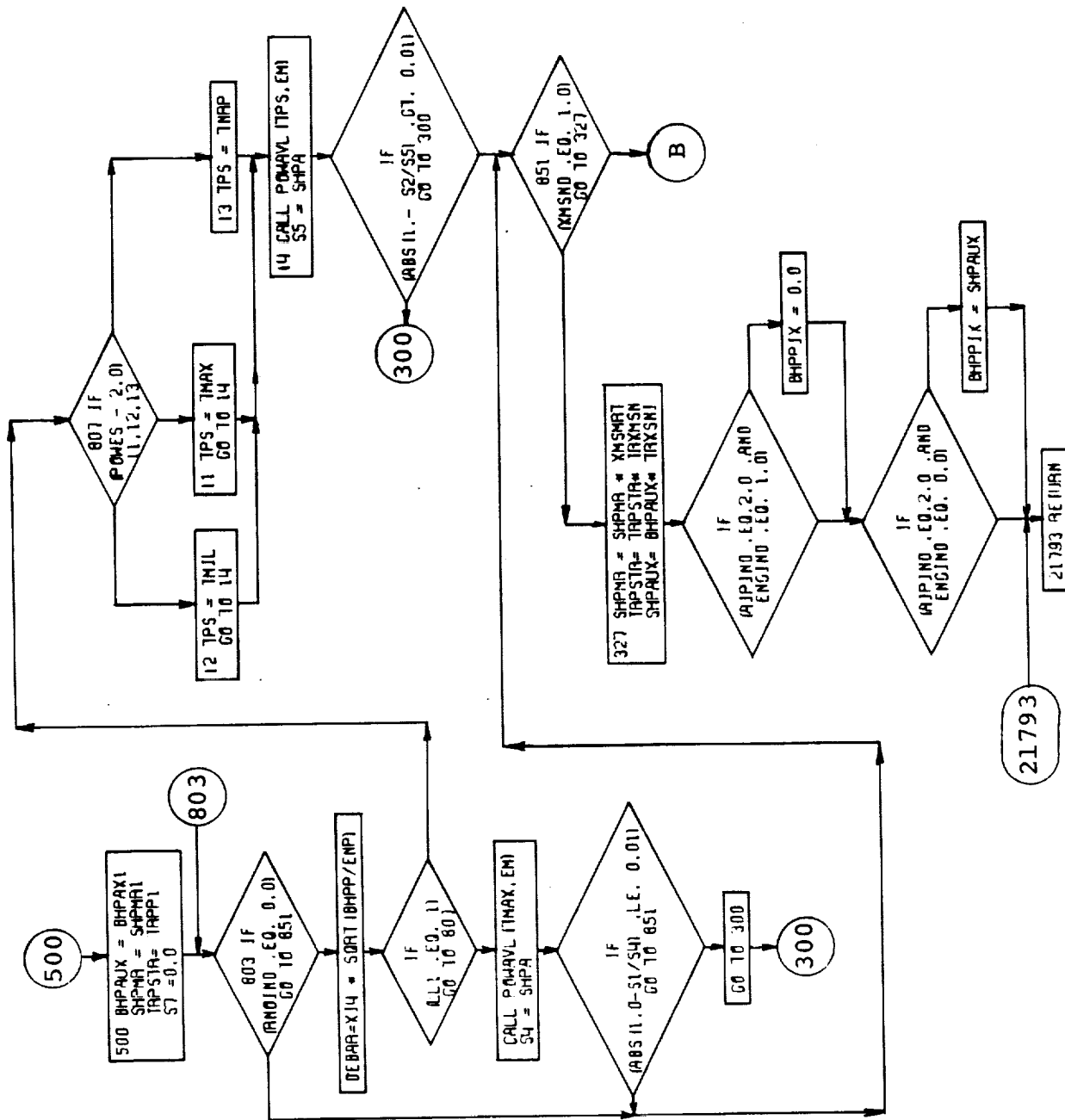


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 8 of 10).

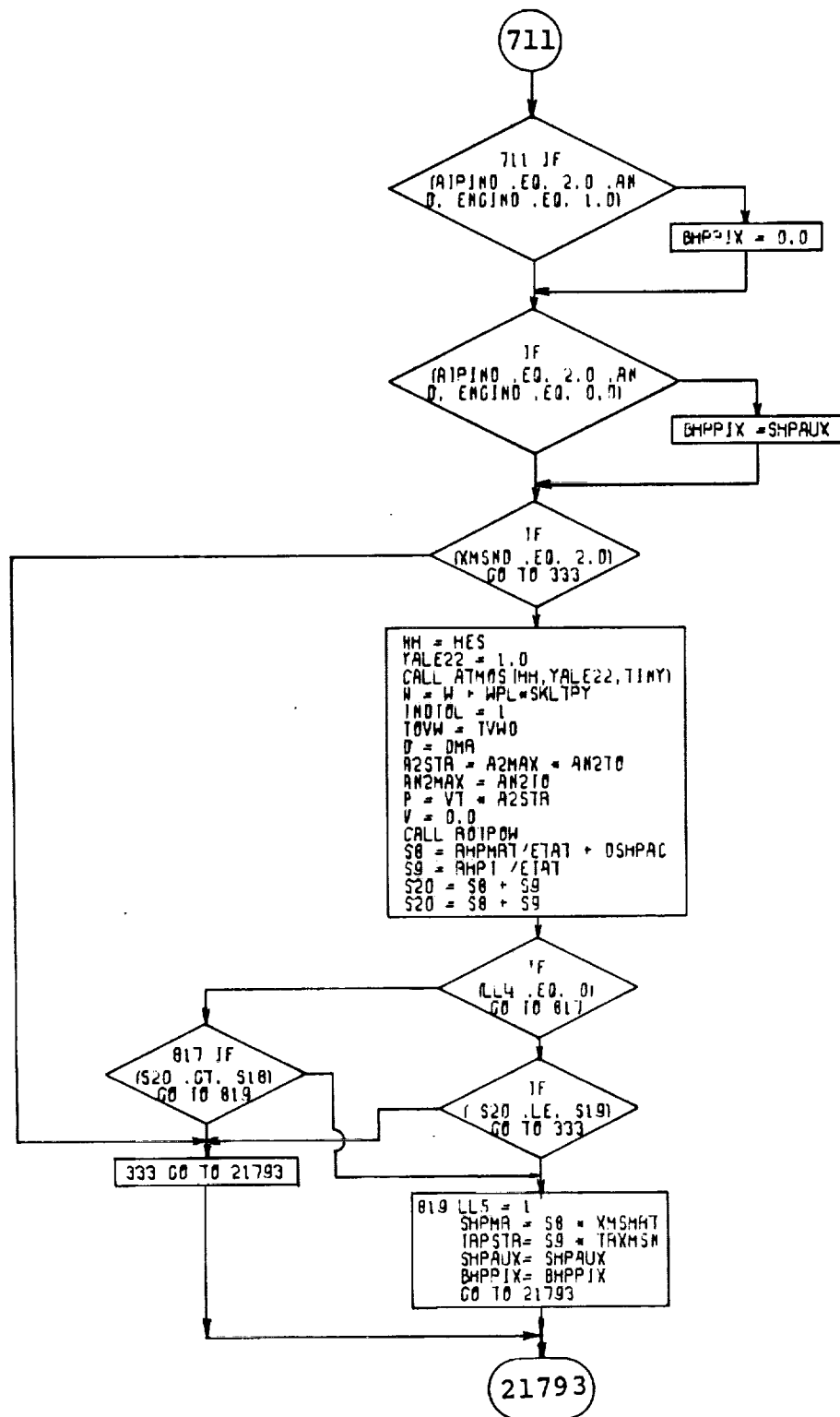


Figure 4-28. Engine Sizing Subroutine Flow Chart
(Part 10 of 10).

4.11 WEIGHT TRENDS SUBROUTINE

The weight trends subroutine calculates the group weights for the propulsion system, the structures system, and the flight control system. These weights are then combined with input values of the weight of fixed useful load, fixed equipment, and payload in order to determine the weight of fuel available (Figure 4-29). The subroutine uses detailed statistical weight equations as used at the Boeing Vertol Company. The group weights are not directly added, but rather are combined by the use of incremental multiplicative and additive weight factors; these factors are useful for sensitivity studies for the aircraft. For example, if it is desired to determine the effect of an additional 300 pounds of propulsion system weight, the factor W_p is input as 300. Similarly, if it is desired to investigate the effect of a 15-percent increase in the weight of the engines, the factor K_{18} is input as 1.15.

In order to calculate the weight of the aircraft structure, the weight trends subroutine must determine the limiting design load factor. For pure and auxiliary propulsion helicopters (without wings), the program uses the input value of maneuver load factor. In the case of a wing or compound helicopter, it does this by comparing the magnitude of the input maneuver load factor with the value calculated for gust load factor. The gust load factor is evaluated at the altitude at which maximum operating equivalent airspeed (V_{MO}) is equal to the speed for maximum operating Mach number (M_{MO}) so long as the altitude falls in the band,

$$0 \leq h_{CRIT} \leq 20,000 \text{ ft}$$

The gust load factor is calculated at the speed V_C (see Reference 11) which is taken to be equal to V_{MO}/M_{MO} . If the user finds that his aircraft is gust-critical at other than the V condition, he must manually calculate the expected load factor and insert that value in the program as a dummy maneuver load factor.

4.11.1 Weight Trend Data

The weights subroutine section of HESCOMP represents one approach for determining the individual and group weights which make up the weight empty of an aircraft. The aircraft weight is divided into subgroups, as shown in Table 4-9, and is in general accordance with the weight and balance data reporting procedures and forms for Aircraft and Rotorcraft described in Military Standard 1374. A copy of Part I (Group Weight Statement) is included at the end of this section. A flow chart describing the weights subroutine is shown in Figure 4-30.

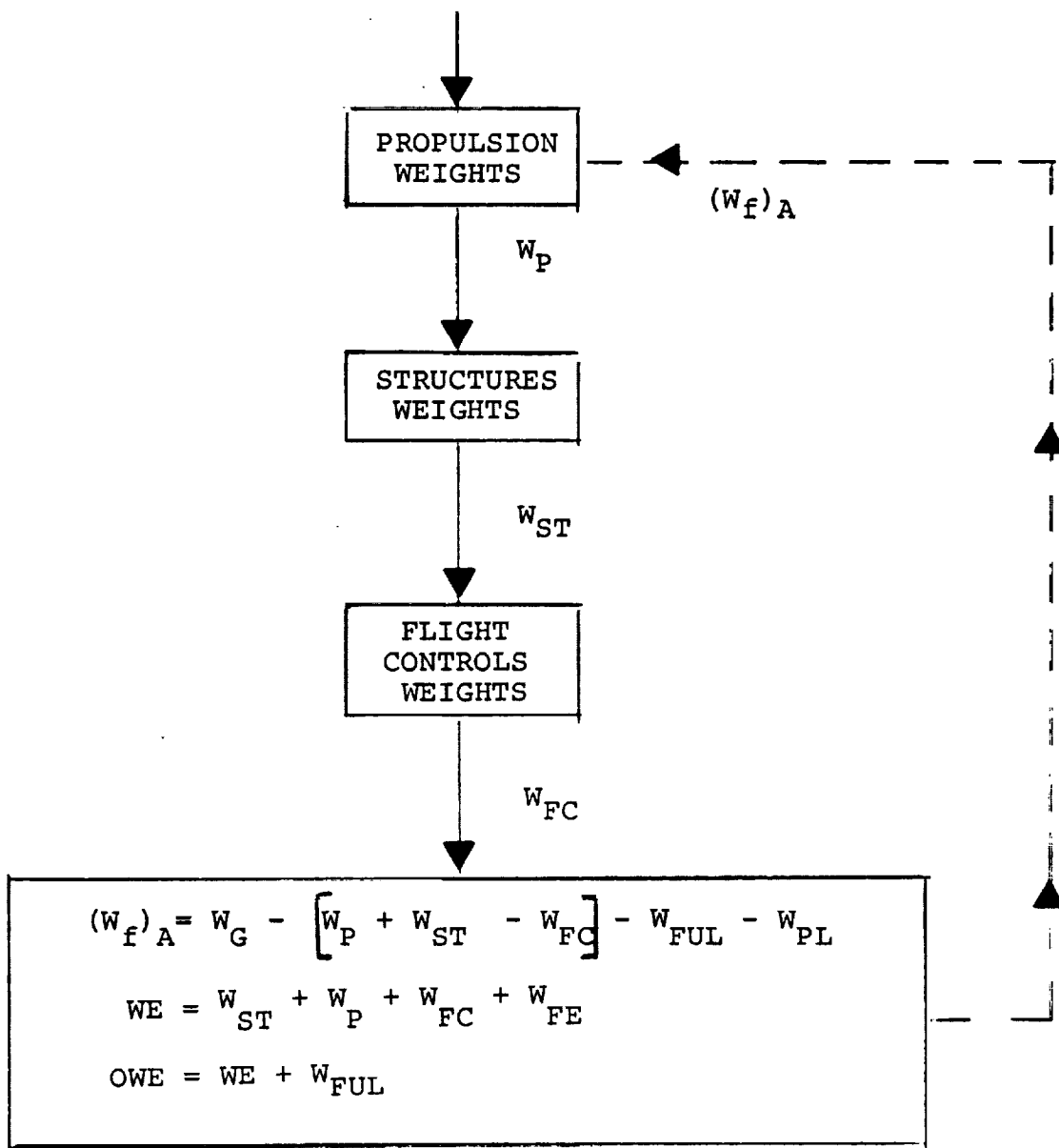


Figure 4-29. Weight Trends Subroutine.

TABLE 4-9. WEIGHT SUMMARY FORM

WING	1			
ROTOR	2			
TAIL	3			
SURFACES	4			
ROTOR	5			
BODY	6			
BASIC	7			
SECONDARY	8			
ALIGNING GEAR GROUP	9			
ENGINE SECTION	10			
	11			
PROPULSION GROUP	12			
ENGINE INST'L	13			
EXHAUST SYSTEM	14			
COOLING	15			
CONTROLS	16			
STARTING	17			
PROPELLER INST'L	18			
LUBRICATING	19			
FUEL	20			
DRIVE	21			
FLIGHT CONTROLS	22			
	23			
AUX. POWER PLANT	24			
INSTRUMENTS	25			
HYDR. & PNEUMATIC	26			
ELECTRICAL GROUP	27			
AVIONICS GROUP	28			
ARMAMENT GROUP	29			
FURN. & EQUIP. GROUP	30			
ACCOM. FOR PERSON.	31			
MISC. EQUIPMENT	32			
FURNISHINGS	33			
EMERG. EQUIPMENT	34			
AIR CONDITIONING	35			
ANTI-ICING GROUP	36			
LOAD AND HANDLING GP.	37			
	38			
	39			
	40			
	41			
WEIGHT EMPTY				
CREW				
TRAPPED LIQUIDS				
ENGINE OIL				
FUEL				
GROSS WEIGHT				

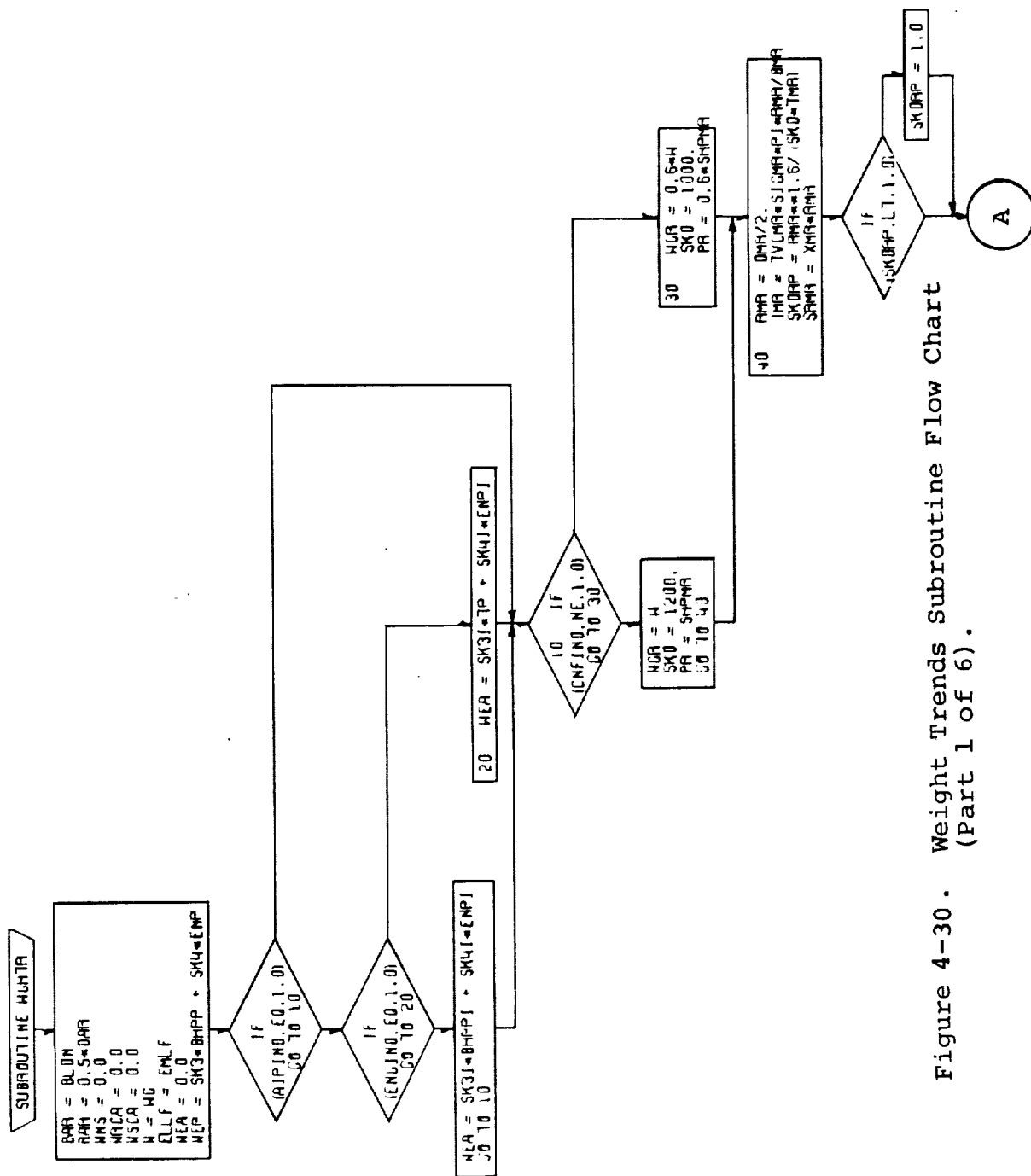


Figure 4-30. Weight Trends Subroutine Flow Chart
(Part 1 of 6).

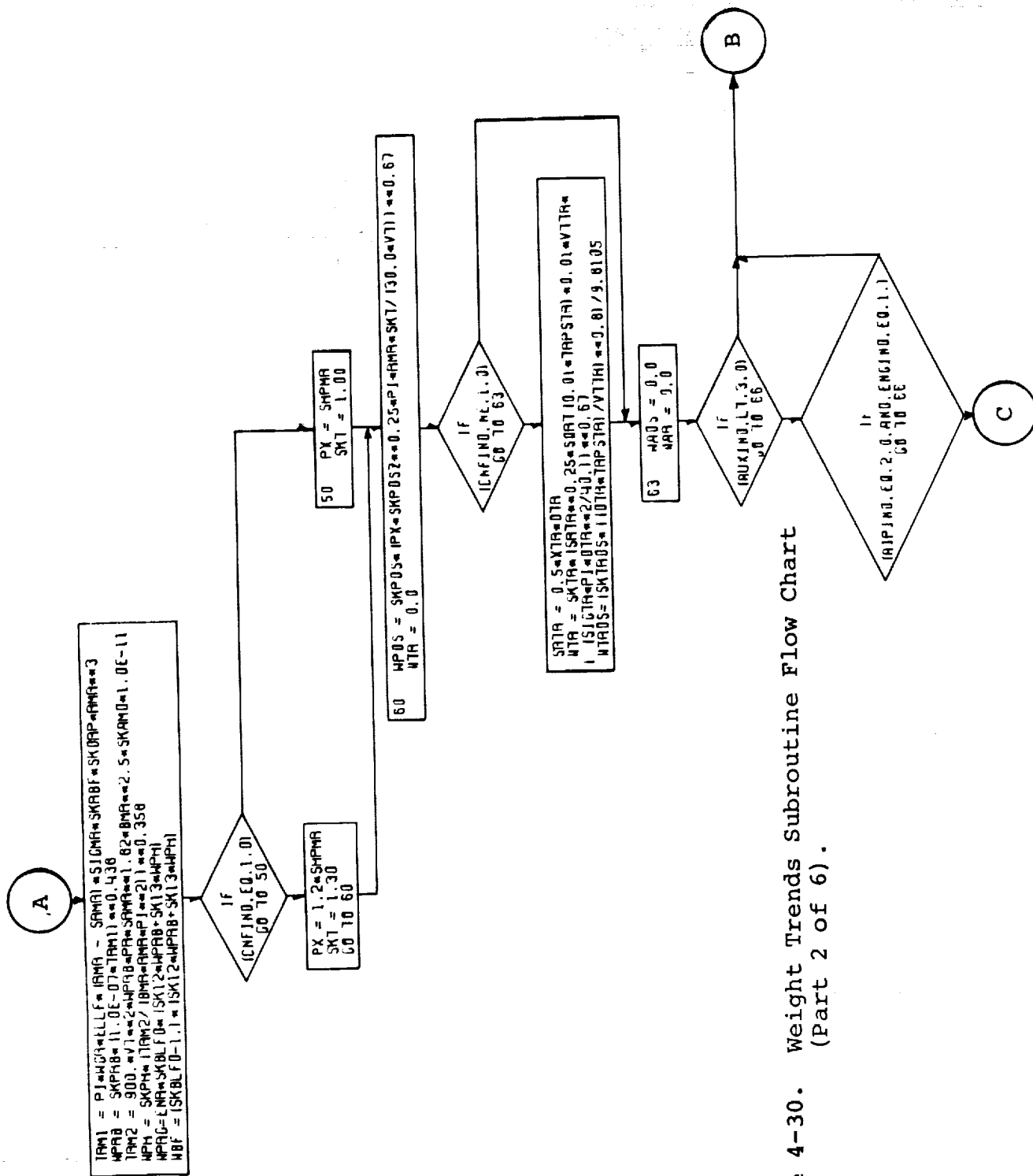


Figure 4-30. Weight Trends Subroutine Flow Chart
(Part 2 of 6).

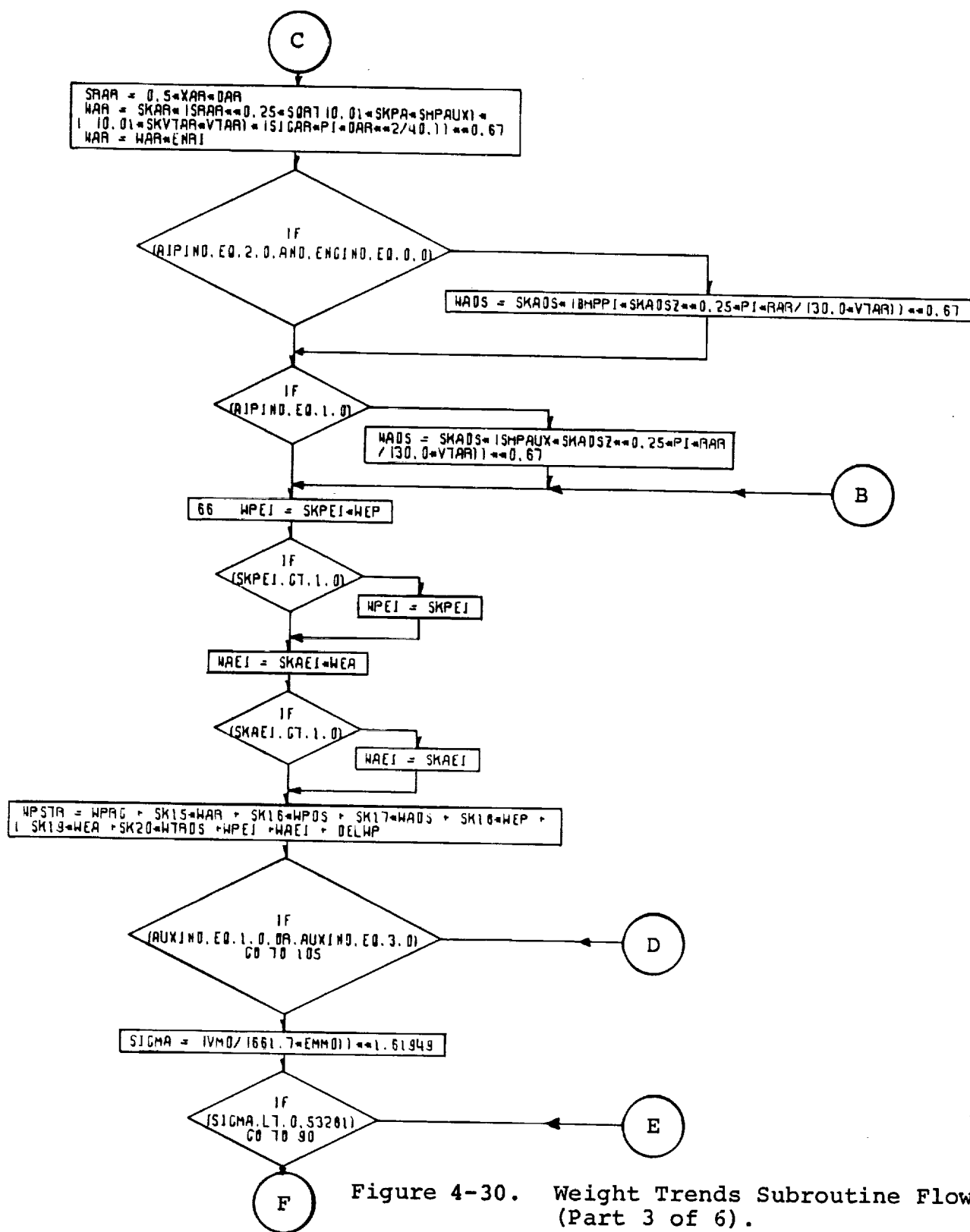


Figure 4-30. Weight Trends Subroutine Flow Chart (Part 3 of 6).

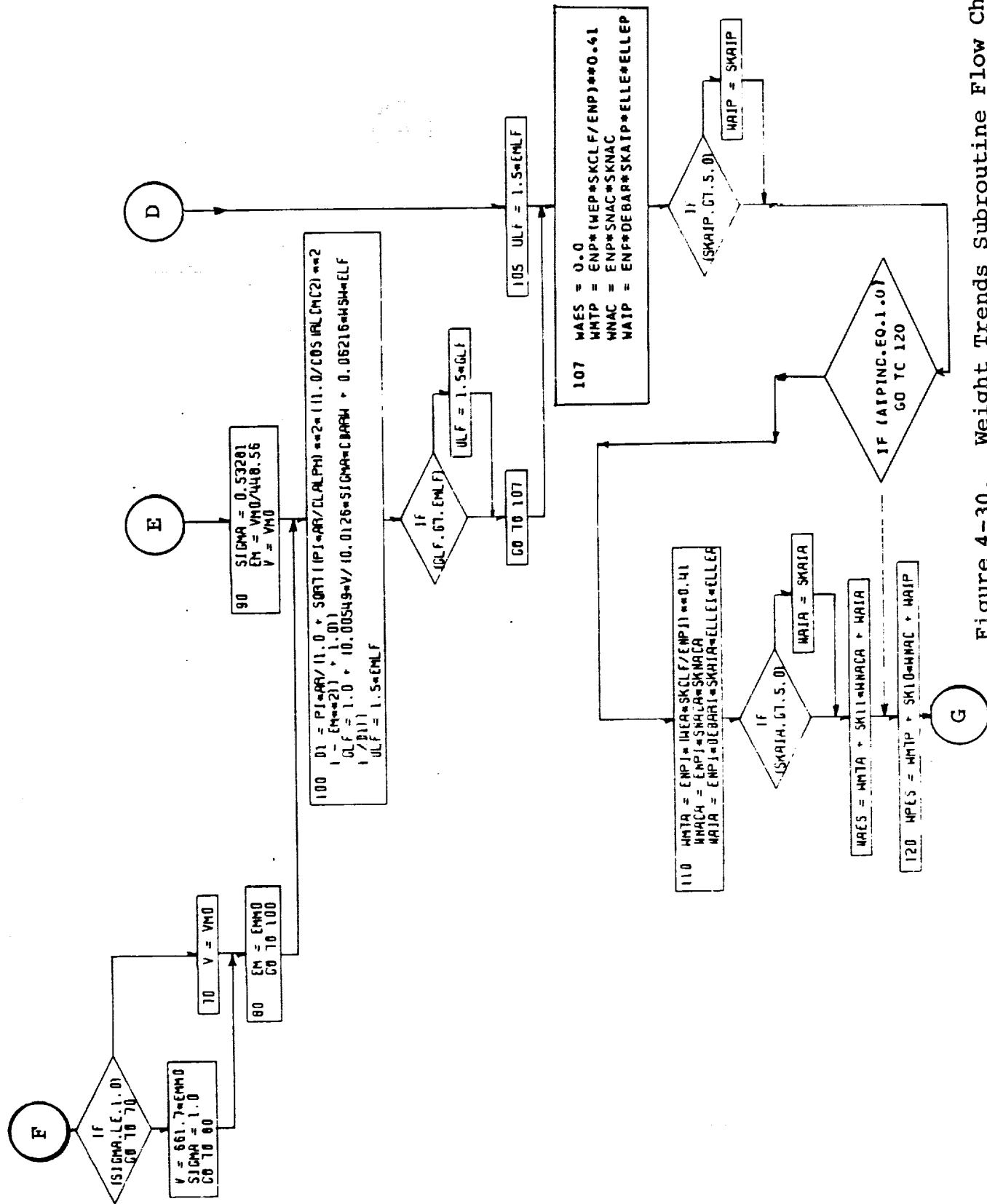


Figure 4-30. Weight Trends Subroutine Flow Chart
(Part 4 of 6).

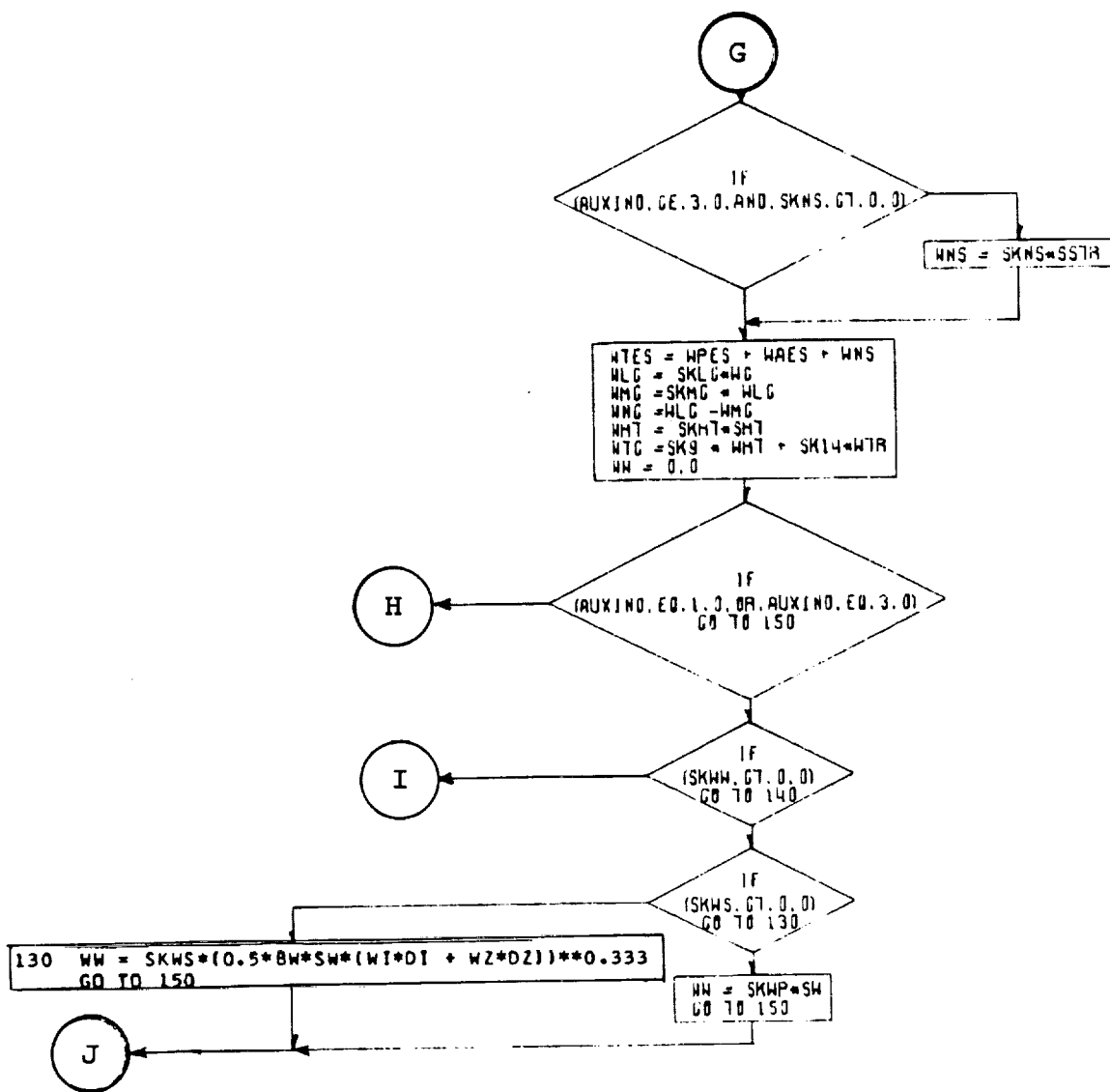


Figure 4-30. Weight Trends Subroutine Flow Chart
(Part 5 of 6).

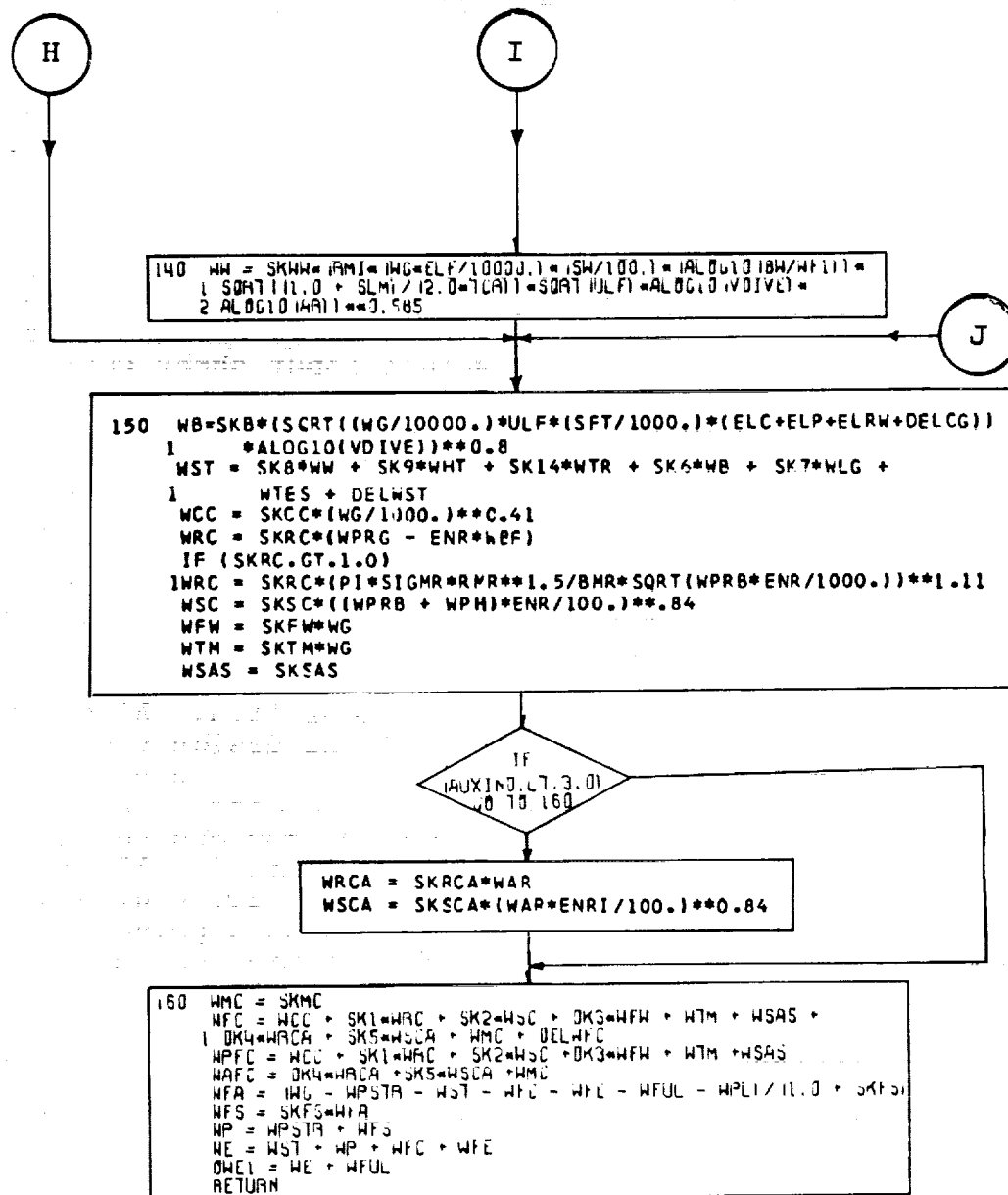


Figure 4-30. Weight Trends Subroutine Flow Chart
(Part 6 of 6).

The trend equations shown on the weights subroutine flow chart and those presented in the text produce the same results, although they are not necessarily written in the same form. The flow chart equations express the text trends in the term used in other parts of the computer program.

The primary purpose of this weights subroutine is to provide a consistent method for rapidly estimating the operational weight empty and fuel available for the missions of various types of helicopters. The results obtained from the trend equations will depend largely on engineering experience and the judgment exercised in selecting the various trend constants. The weight trend equations were developed by A. H. Schmidt and R. H. Swan of Boeing Vertol Company.

An explanation of the weight trends and instructions for completing the weight input sheet are included in the text. As an additional aid for filling out the weight input sheet, the page numbers defining the various k terms are included with the respective terms on the weight input sheet (Table 10).

Weight trends developed at Boeing were used to determine the structure weights, Table 4-9, items 1, 6 and 10; flight control weights, item 22; and the control and propulsion system weights, items 2, 5, 18 and 21. The trends were developed from existing aircraft, and use design and geometric parameters to compute the weights of the various components. For aircraft on which limited information is available, such as compound, winged and propulsive tail helicopters, the trend constants have been adjusted to account for the design features typical of the particular configuration. Alighting gear weights, item 9, are a function of the design takeoff weight and are based on statistically derived percentages of the respective gross weights. Engine weights, item 13, were determined from information compiled from engine manufacturers. Engine installation weights, items 14, 15, 16, 17 and 19, are expressed as a percentage of the dry engine weight. Fuel system weight, item 20, is determined on a pound per gallon of fuel required basis. Fixed equipment weights, items 24 through 41, are discussed in the text.

Table 4-9 is representative of a typical weight summary form used for military aircraft. Weight definitions as used in MIL-STD-1374 and weight handbooks follow.

6.2.3 Weight Definitions - As used in MIL-STD-1374 and Weight Handbooks.

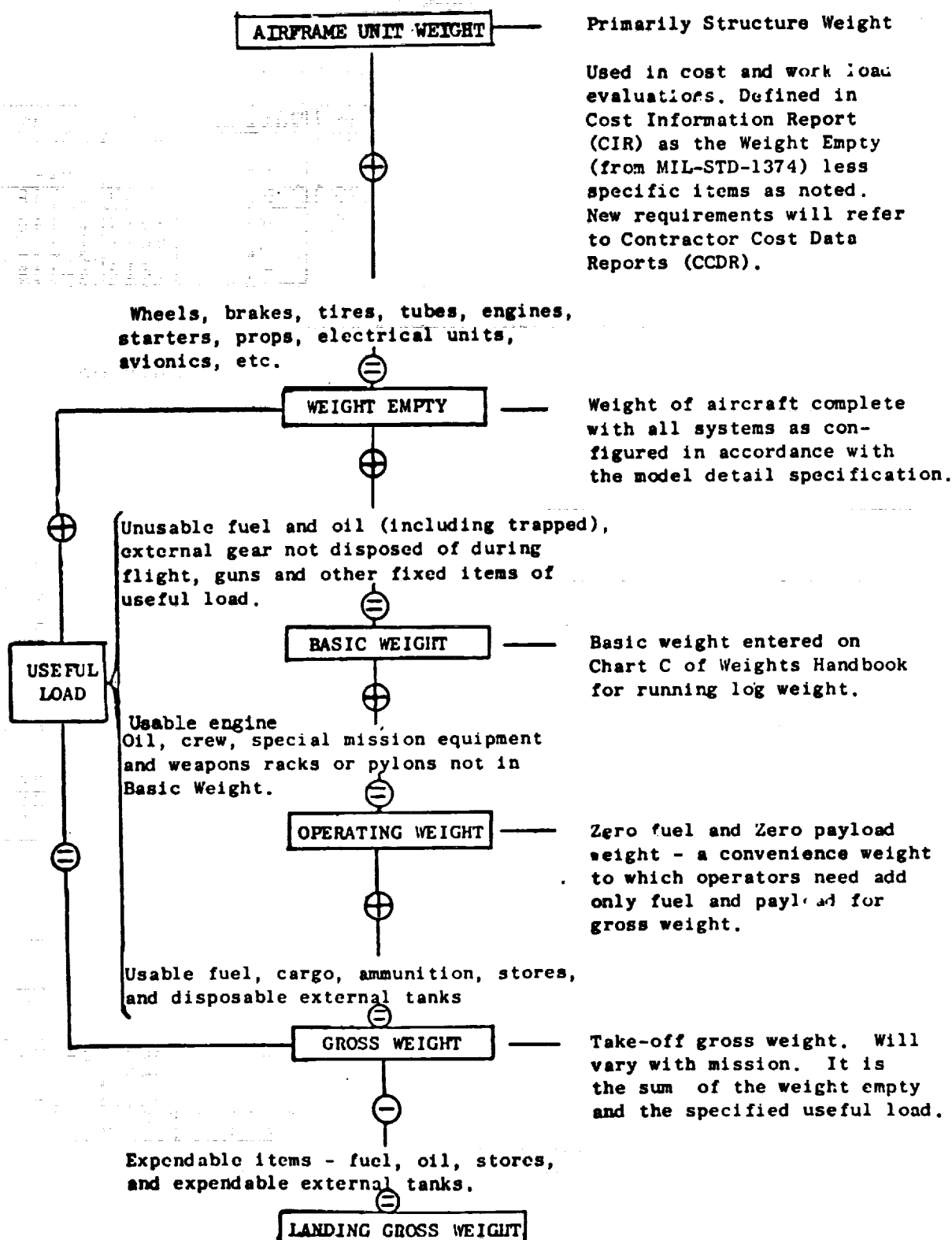


TABLE 10. HELICOPTER WEIGHT INFORMATION

Incremental Group Wts Nom = 0

Variable	LOC	Value	Variable	LOC	Value	Variable	LOC	Value
OWE	2601	4-136	ΔW_{FC}	2605	4-175	RM _I	2608	4-147
WFE	2602	4-172*	ΔW_P	2606	4-175	W _i	2609	4-148
WFUL	2603	4-172	ΔW_{ST}	2607	4-175	W _o	2610	4-148
WPL	2604	4-175				L _i	2611	4-148
						L _o	2612	4-148

Group Weight Information

Flight Controls

kCC	2613	4-168
kRC	2614	4-168
kSC	2615	4-168
kFW	2616	4-168
kTM	2617	4-168
KSAS	2618	4-172
kRCA	2619	4-172
kSCA	2620	4-172
kMISC	2621	4-172

† OWE IS not
necessary when
OPTIN = 1, 2

†† W_{PL} is not
necessary when
OPTIN = 2

Structural

kB	2622	4-156
$\Delta C.G.$	2623	4-156
kLG	2624	4-158
kMG	2625	4-158
kWW	2626	4-147
LF	2627	4-147
kWS	2628	4-148
kWP	2629	4-148
kHT	2630	4-153
kCLE	2631	4-160
KNAC	2632	4-160
KAIP	2633	4-160
KNACA	2634	4-160
KATA	2635	4-160
KNS	2636	

Propulsion

kPRB	2637	4-150
kPRF	2638	4-150
kPH	2639	4-150
kamd	2640	4-150
kBLFD	2641	4-150
kTR	2642	4-155
kAR	2643	4-153
kPA	2644	4-153
kVTAR	2645	4-153
kpDS	2646	4-164
kpDSZ	2647	4-164
kTRDS	2648	4-166
kADS	2649	4-164
kADSZ	2650	4-164
kFS	2651	4-162
kPEI	2652	4-161
kAEI	2653	4-161

k ₁	2654	4-175
k ₂	2655	4-175
k ₃	2656	4-175
k ₄	2657	4-175
k ₅	2658	4-175

k ₆	2659	4-175
k ₇	2660	4-175
k ₈	2661	4-175
k ₉	2662	4-175
k ₁₀	2663	4-175
k ₁₁	2664	4-175

k ₁₂	2665	4-175
k ₁₃	2666	4-175
k ₁₄	2667	4-175
k ₁₅	2668	4-175
k ₁₆	2669	4-175
k ₁₇	2670	4-175
k ₁₈	2671	4-175
k ₁₉	2672	4-175
k ₂₀	2673	4-175

*Page numbers in this document

MULTIPLICATIVE FACTORS
NOMINALLY = 1.0

Wing

The weight of the wing is determined using one of the following three methods:

● Method I

$$W_W = 220 (k)^{0.585},$$

Where

$$k = \left[\frac{R_M}{10^4} \right] \left[\frac{W_g LF}{10^4} \right] \left[\frac{S_W}{10^2} \right] \left[\log \frac{b}{B} \right] \left[\sqrt{\frac{1+\lambda}{2K}} \right] \left[\sqrt{N} \right] \left[\log_{10} V_D \right] \left[\log_{10} A_R \right]$$

Legend

W_W = weight of wing - lb

R_M = wing relief as a fraction of design gross weight

W_g = design gross weight

LF = helicopter lift factor as a fraction of gross weight

S_W = planform area of wing (taken from C_L of aircraft) - ft²

b = wingspan - ft

B = maximum fuselage width - ft

λ = taper ratio

N = ultimate load factor

V_D = dive velocity - kn

A_R = aspect ratio

k_r = wing root thickness $\div$ root chord

Method I is used when a conventional aircraft wing is employed. It considers basic geometry, design criteria, and relief terms. The 220 constant represents a wing employing simple control surfaces. The 220 adjusted up or down depending on the complexity of the surface controls (200-240) must be placed in the k_{ww} location on the weight input sheet. LF , representing the wing unloading factor, due to rotor lift, and R_M a wing relief value (0.5 to 0.75) must be entered as a fraction of the design gross weight. The factors LF and R_M are nominally 1.0.

● Method II

$$W_W = 3.15(k)^{0.333}$$

Where

$$K = S_W (W_i L_i + W_o L_o)$$

Legend

W_W = weight of wing - lb

S_W = planform area of wing - ft^2

W_i = inboard wing store weight, lb/per side

W_o = outboard wing store weight, lb/per side

L_i = distance from side of fuselage to inboard store - ft

L_o = distance from side of fuselage to outboard store - ft

Method II is used when a sponson or stub type wing is used to carry stores or weapons. The trend constant 3.15 must be placed in k_{ws} of the weight input sheet. W_i and W_o must be entered in their respective locations in pounds per side. L_i and L_o must be entered as a fractional part of the wing semi-span.

● Method III

$$W_W = S_W \times PSF$$

Where

W_W = weight of wing - lb

S_W = planform area of wing - ft^2

PSF = pounds per ft^2

Method III is used when a single sponson or stub is employed. The estimated unit weight of the wing in pounds per square foot is placed in k_{wp} on the weight input sheet.

Main Rotors

The weight of the main rotor includes the combined weights of the blades and hub and hinge. The weights are derived from the following equations:

- Blades (per rotor) $W_B = 44 a (k)^{0.438}$

Where

$$k = \left[\frac{W_g}{10^4} \right] [LLF] \left[\frac{R^2}{100} \right] \left[\frac{R-r}{10} \right] \left[\frac{b}{c} \right]^{k_b} \left[\frac{R}{k_d t} \right]^{1.6}$$

NOTE: The last term is a droop factor, used only if the result is greater than 1.

- Hub and Hinge (per rotor) $W_{HH} = 61 a (k)^{0.358}$

Where

$$k = \left[\frac{W_B}{10^4} \right] \left[\frac{R}{10} \right] \left[\frac{N_R}{10} \right]^2 \left[\frac{P_R}{10} \right] \left[\frac{r}{10} \right]^{1.82} \left[\frac{b}{10} \right]^{2.5} \left[k_{amd} \times 10^{-11} \right]$$

Legend

W_B = blade weight per rotor (including root end fitting)-lb

a = adjustment factor

W_g = design gross weight per rotor (X 0.6 for tandem)-lb

LLF = design limit load factor at dgw

R = rotor radius-ft

r = rotation to blade attachment-ft

c = blade chord-ft

b = number of blades per rotor

t = maximum blade thickness at 25% R-ft

k_b = rotor type factor: 1.00 articulated, 2.2 hingeless or teetering

k_d = droop constant: 1000 tandem, 1200 single rotor

W_b = blade weight per blade (including root end fitting)-lb

N_r = rotor rpm

P_r = takeoff power X (0.6 for tandem) per rotor-hp

k_{amd} = a x m x d

a = design concept: 0.53 hingeless, 1.00 other

m = material: steel = 1.00, titanium = 0.54

d = development stage: early = 1.0, developed = 0.62

In the trend equations the constants 44(blade trend) and 61 (hub and hinge trend) represent the average for the rotor weights presented in Figure 4-31 and 4-32. The blade weights are most representative of the all metal blades. The adjustment factor a is used to adjust the k factor when special design features are considered, such as high modulus materials (boron, graphite, etc.) or special features associated with the hub and/or hinge. Refer to Figures 4-31 and 4-32 to select the a term which most closely approximates the configuration being analyzed. The revised constants 44a and 61a must be placed in the k_{PRB} and k_{PH} locations on the weight input sheet along with the factors k_{RBF} (k_b in legend) and k_{amd} . If blade folding is required the k_{BLFD} block on the input sheet must also be filled in. Blade folding is entered as a fractional part of the total rotor weight. The blade fold penalty usually runs between 0.15 to 0.25 of the rotor weight depending on the folding requirements. The nominal value for k_{BLFD} is 1.0.

Auxiliary Rotors or Propellers

When auxiliary rotors or propellers are required, as in the case of compounds or propulsive tail helicopters, the following rotor/propeller equation is used:

$$W_R = 14.2 a (k)^{0.67},$$

Where

$$k = [r]^{0.25} \left[\frac{H_{pr}}{100} \right]^{0.5} \left[\frac{V_{tl}}{100} \right] \left[\frac{R.b.c.}{10} \right]$$

Legend

W_R = weight of rotor or propeller-lb

R = rotor radius-ft

b = number of blades per rotor

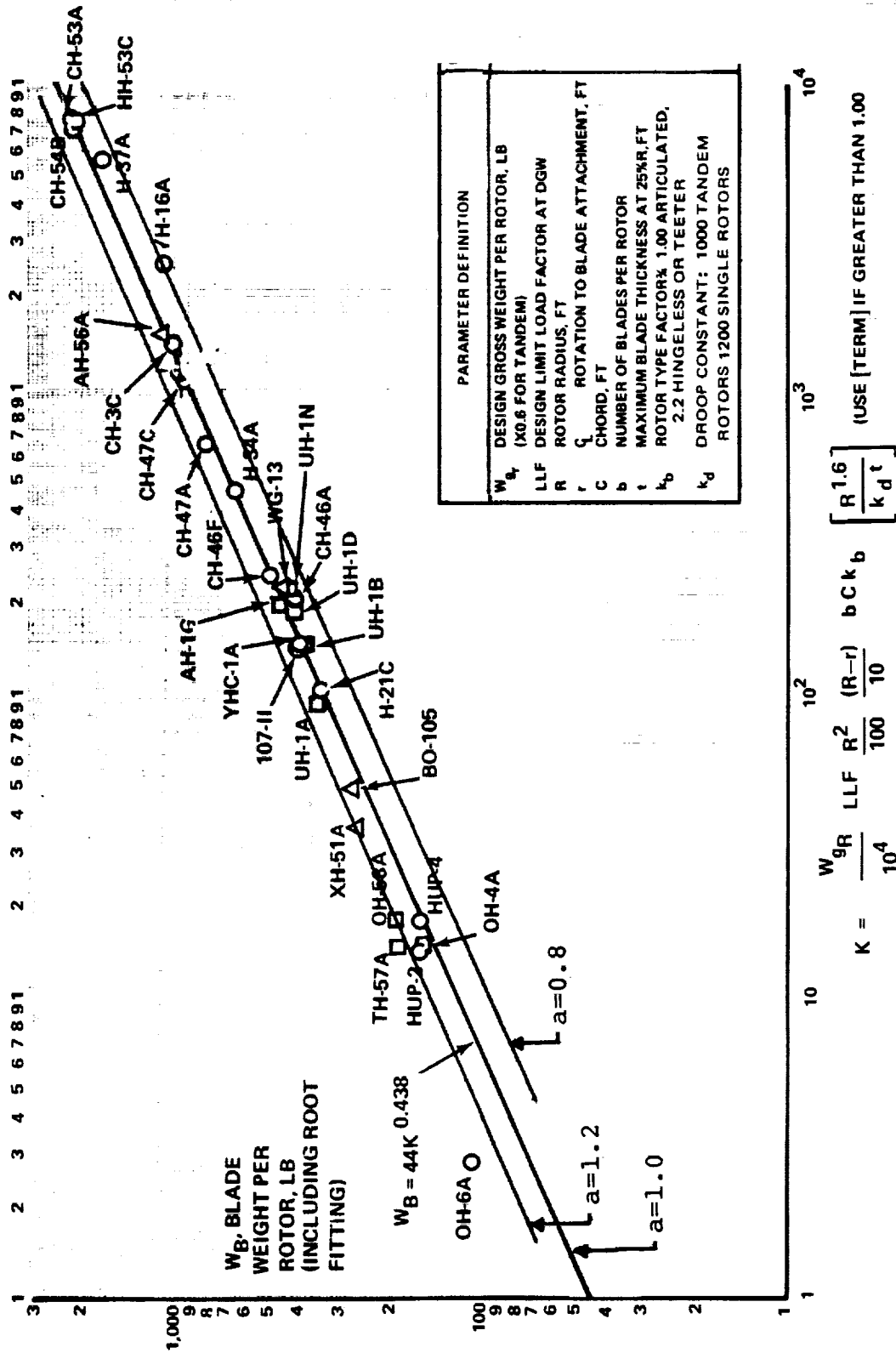


Figure 4-31. Rotor Blade Weight Trend.

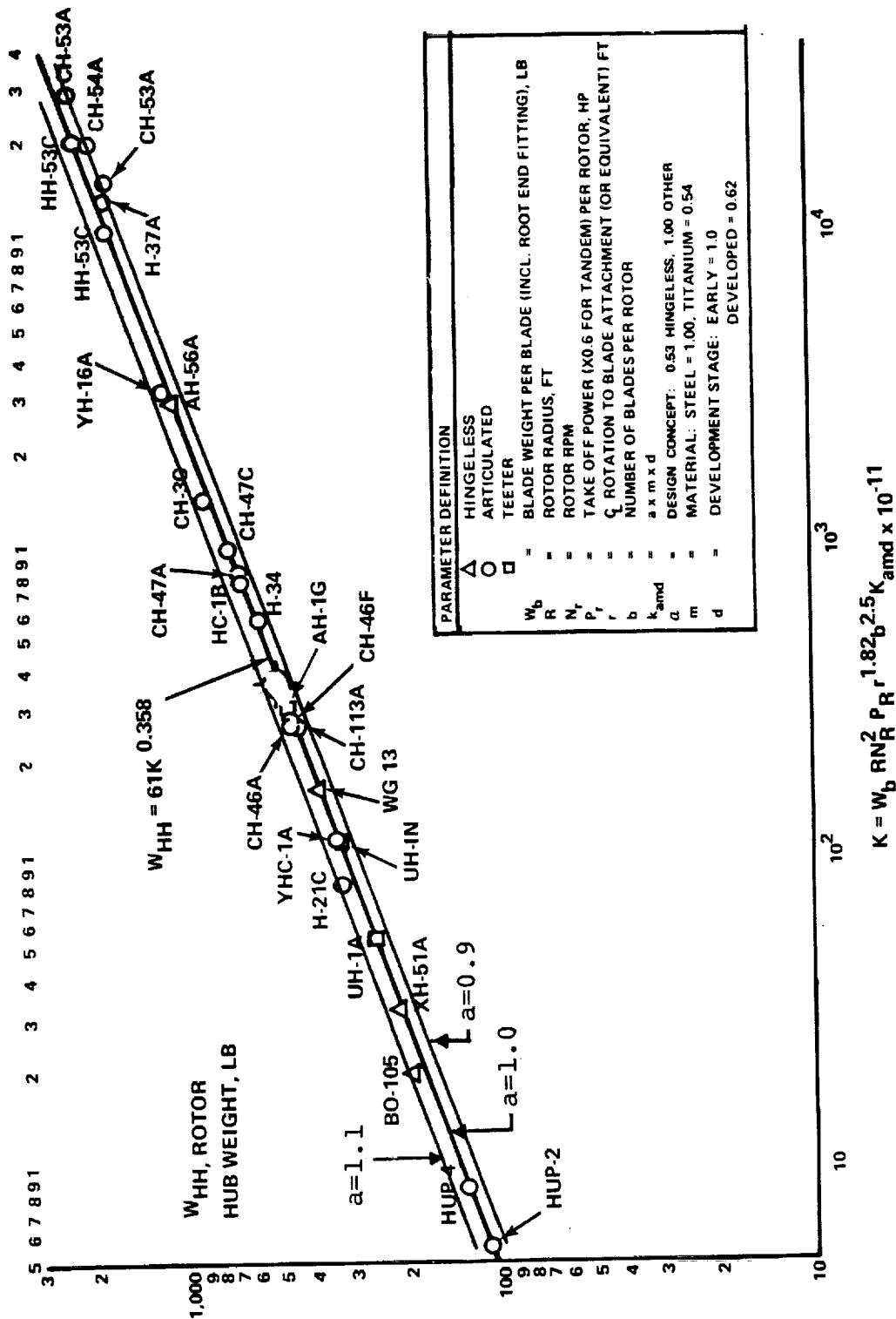


Figure 4-32. Rotor Hub and Hinge Weight Trend.

c = blade chord (average)-ft

HP_r = horsepower (xmsn limit per rotor)

V_{tl} = design limit tip speed-ft/sec

r = center line of rotation to average blade attachment point-ft

a = adjusting factor for type of system (see Figure 4-33)

In the trend equation the constant 14.2 is the average for the various rotor group weights presented in Figure 4-33. The expression a is the adjustment factor for the type of system; i.e., semirigid, pressure cycle, etc. To determine the value of k_{AR} in the propulsion block of the weight input sheet, multiply the type of system desired a by the constant 14.2. Blade folding, if required, is entered in k_{15} as a percentage factor of the total computed rotor weight. The input value would be between 0.15 to 0.25 depending on the folding requirements.

The k_{PA} block on the weight input sheet allows the auxiliary rotor input power to be increased or decreased as a fractional part of input power. ($k_{PA} = 0.9$ would decrease the power by 10 percent, 1.1 would increase the power by 10 percent.)

The k_{VTAR} block on the weight input sheet allows the auxiliary tail rotor tip speed to be increased or decreased as a fractional part of the input tip speed ($k_{VTAR} = 0.9$ would decrease the tip speed by 10 percent, 1.1 would increase it by 10 percent). The nominal input values for k_{PA} and k_{VTAR} is 1.0.

Tail Rotor

The tail group consists of the horizontal tail, vertical tail, ventral and tail rotor. Tail weights are determined as follows:

- Horizontal Tail - Its weight is based on a unit weight per square foot (PSF). The unit weight will normally vary between 1.0 and 2.0 PSF, depending on the type of tail being employed. The unit weights of the horizontal tails of some existing helicopters are presented as a guide for inputting the unit value in the k_{HT} block of the weight input sheet.

OH - 58A 1.1 lb/ft² - fixed

UH - 1H 1.3 lb/ft² - movable

UH - IN 1.6 lb/ft² - stabilizer

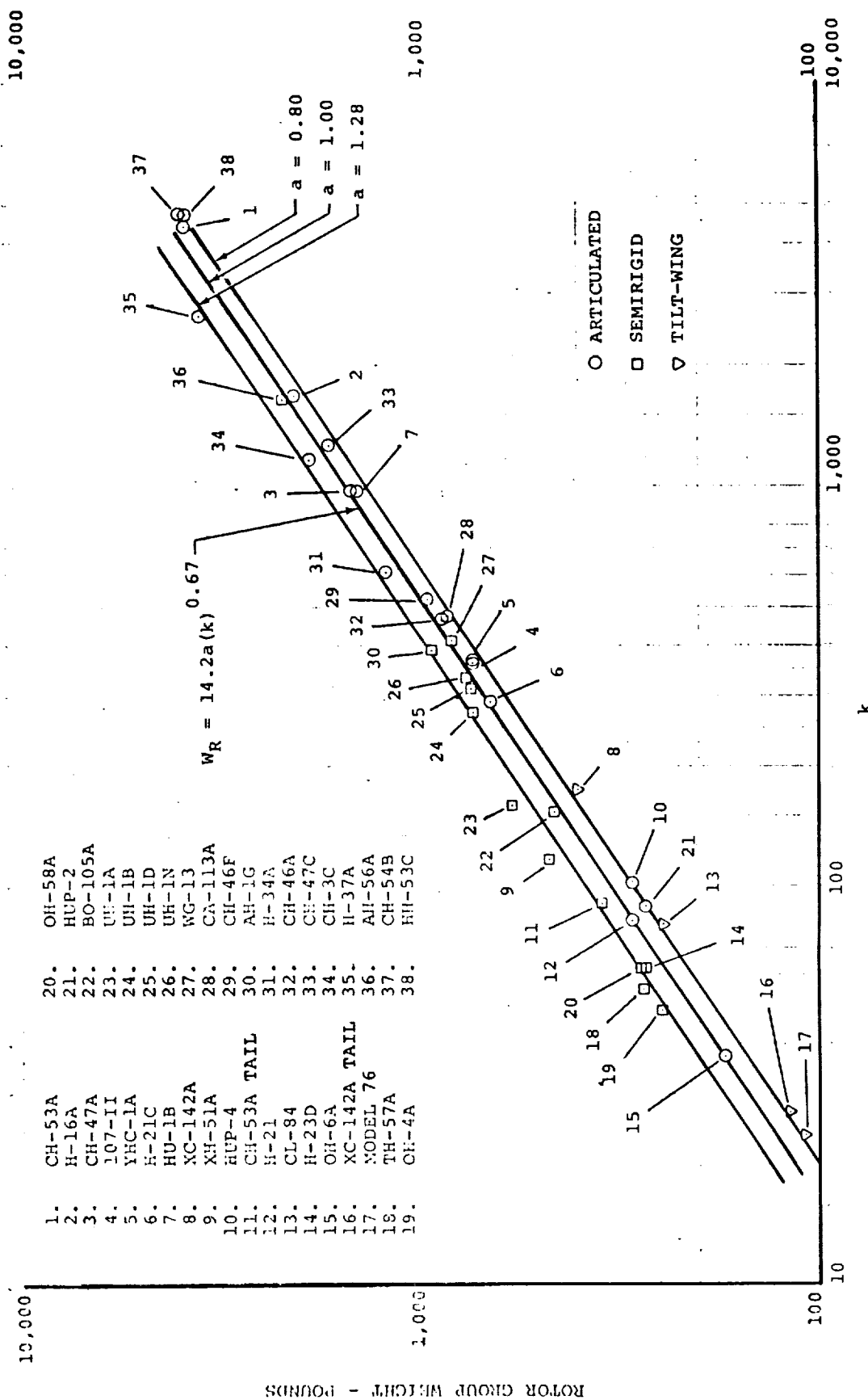


Figure 4-33. Rotor Group Weight Trend.

If horizontal tail fold is required, input this as a percentage of the total horizontal tail weight in the k_9 block of the input sheet (refer to Table 4-13).

- Vertical Tail and Ventral - The combined weights of vertical tail and ventral are included in the weight of the fuselage. The combined wetted area of both must be added to the fuselage wetted area.
- Tail Rotor - The weight of the tail rotor is derived from the following overall rotor trend equation:

$$W_R = 14.2 a (k)^{0.67},$$

Where

$$k = \left[r \right]^{0.25} \left[\frac{HP_r}{100} \right]^{0.5} \left[\frac{V_{t1}}{100} \right] \left[\frac{R.b.c.}{10} \right]$$

Legend

W_R = weight of rotor or propeller-lb

R = rotor radius-ft

b = number of blades per rotor

c = blade chord (average)-ft

HP_r = horsepower (xmsn limit per rotor)

V_{t1} = design limit tip speed-ft/sec

r = center line of rotation to average blade attachment point-ft

a = adjusting factor for type of system (see Figure 4-33)

This is the same equation used to determine the weight of the auxiliary rotors. The trend is explained above under Auxiliary Rotors or Propellers. If blade folding is required, a factor as a fraction of the computed tail rotor weight must be placed in k_{14} on the weight input sheet. Fold penalties normally vary between 0.15 and 0.25 of total rotor weight depending on the fold requirements. A value for k_{TR} must be inserted in its proper location on the weight input sheet.

Body Group

The weight of the body structure is determined from the following equation:

$$W_{BG} = 125 a (k)^{0.8},$$

Where

$$k = \left[\frac{W_g}{10^4} \right] \left[N \right] \left[\frac{S_f}{10^3} \right] \left[L_c + L_{rw} + CG \right]^{0.5} \left[\log V_{MAX} \right]$$

Legend

W_g = structural design gross weight - lb

N = ultimate load factor

S_f = wetted area of fuselage - ft² (includes fairings, pod, vertical tail and ventral)

L_c = length of cabin (measured from nose to end of cabin floor) - ft

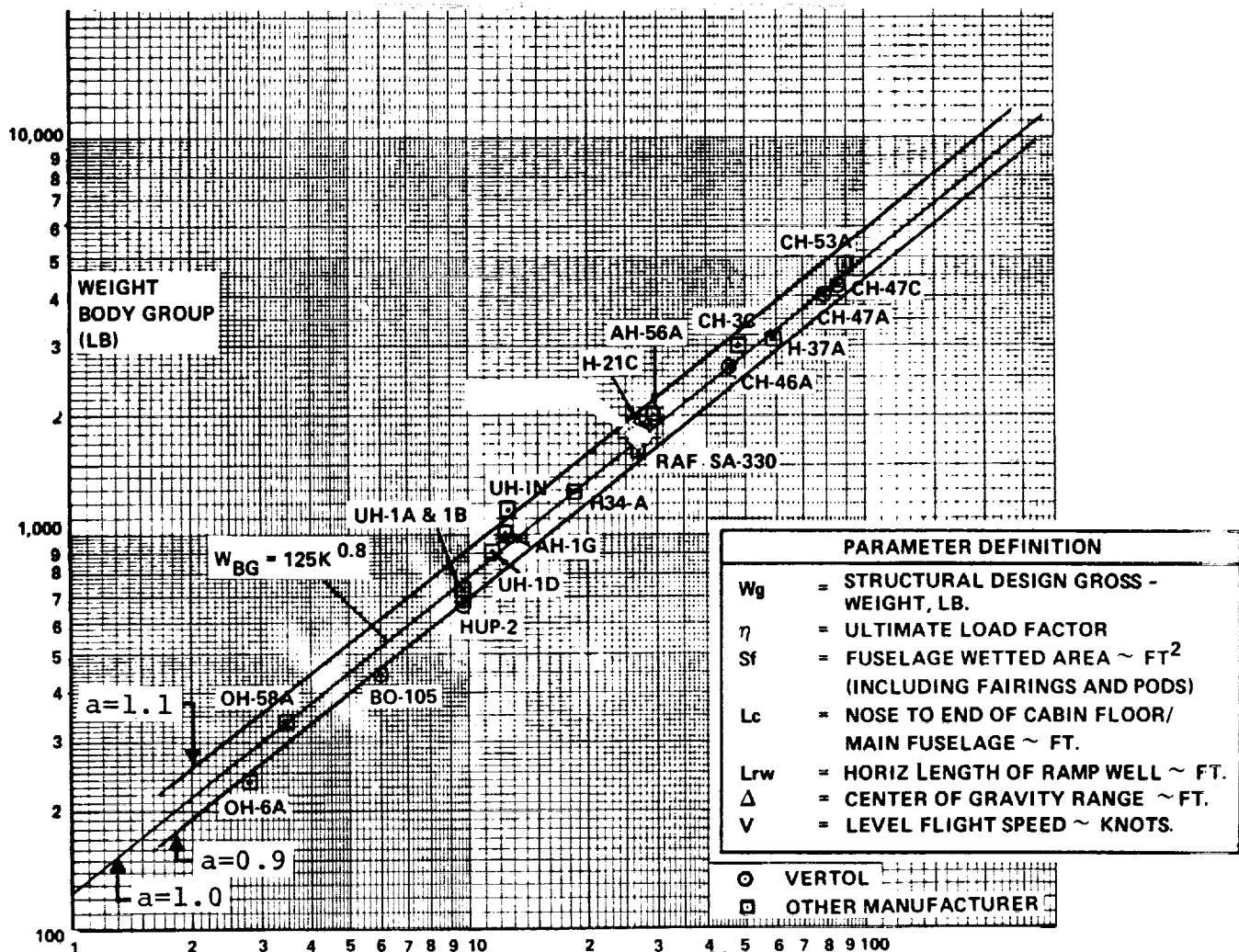
L_{rw} = length of rampwell - ft

CG = center of gravity range at design gross weight - ft

V_{MAX} = maximum speed - kn

a = body correction factor

Figure 4-34 presents a group of commercial and military single and tandem rotor helicopters. A mean line of 125 has been selected as the average for all the aircraft shown. The body correction factor a permits the 125 constant to be corrected in accordance with the configuration being analyzed. When a large number of cutouts are required as in the case of large doors, many windows, large floor cutouts, etc., the a term would be greater than 1. Where the fuselage is relatively clean, the a could be less than 1. Refer to Figure 4-34 to select the a term which best describes the configuration. The revised constant 125a is the k_B term to be inserted in the appropriate box on the weight input sheet. The center of gravity range, in feet, must also be placed in the Δ_{CG} block on the input sheet.



$$K = \frac{W_g}{10^4} \eta \frac{S_f}{10^3} (L_c + L_{rw} + \Delta)^{0.5} \log V_{max}$$

Figure 4-34. Body Group Weight Trend.

Alighting Gear

For the normal tricycle gear geometry, the total landing gear weight including the running gear (wheels, tires, brakes, etc.), structure (shock struts, drag struts, support structure, etc.), and controls (retraction, steering, systems, etc.) is expressed as a percentage of the design gross weight where:

$$W_{LG} = (k_{LG}) W_g$$

Where W_{LG} = total weight of the landing gear (including tail bumper)

$$k_{LG} = \frac{\text{landing gear}}{\text{gross weight}}$$

W_g = design gross weight

The percentage will normally vary between 0.015 to 0.050 depending on the design limit sink speed and the complexity of the system. Conventional landing gear without retraction, operating on improved runways normally run between 0.015 to 0.04. Adding retraction usually adds another 0.005 to 0.01. Skid type landing gear usually weigh about 0.015 times design gross weight.

The main gear usually weighs about 80 percent of the total gear weight. The k term in the weight expression above is the value that must be placed in the k_{LG} box of the weight input sheet. The weight of the main gear is included by placing 0.80, or your estimate of the main gear weight as a fraction of the total gear weight, in the k_{MG} location on the weight input sheet.

Table 4-11 is included as a guide in selecting k_{LG} . It includes the total gear weight as a percentage of the gross weight for a sampling of helicopters.

Engine Section (Primary and Auxiliary)

The engine section weight appears as item 10 in Table 4-9. It is basically the engine mounts, engine nacelle structure and firewalls, air induction and support structure.

- Engine mounts - The weight of the engine mounts is determined from the expression

$$W_{EM} = N_E \left(W_E \times N_{CLF} \right)^{0.41}$$

TABLE 4-11. LANDING GEAR WEIGHTS

AIRCRAFT	GROSS WEIGHT (LB)	TOTAL GEAR WEIGHT (LB)	PERCENT OF GROSS WEIGHT
OH-6A	2400	58	.024
XH-51A	3500	134	.038
BO-105	4410	96	.022
UH-1B	6600	112	.017
UH-1D	6600	118	.018
UH-1N	10000	121	.012
UH-34D	11291	413	.036
AH-56A	16995	605	.036
CH-46A	19000	589	.031
CH-3C	19500	690	.035
CH-46D	20800	587	.028
CH-46E	20800	655	.031
HH-3B	21187	700	.033
CH-47A	28550	1060	.037
CH-47B	33000	1086	.033
CH-47C	33000	1076	.033
CH-53A	35042	1014	.029
CH-54A	38000	1794	.047
CH-54B	64700	2277	.035

Enter the crash load factor N_{CLF} in the k_{CLF} block of the weight input block. (This considers both the primary and/or auxiliary engine sections.)

- Engine nacelle structure, supports and firewalls

These items are determined from the expression

$$W_{NAC} = N_E (S_{NAC}) \text{ (PSF)}$$

Enter the estimated unit weight in pounds per square foot (PSF) in the k_{NAC} and/or the k_{NACA} blocks of the input sheet. This value normally varies between 0.75 and 1.25 PSF. It could go as high as 2.0 psf if the cowling is used as a walkway or work platform.

- Air induction - The weight of the air induction system is determined from the expression

$$W_{AIS} = N_E (E_{DIA} \times L_{AID}) \text{ (PSF)}$$

Enter the estimated unit weight in pounds per square foot (PSF) in the k_{AIP} and/or the k_{AIA} blocks of the input sheet. This value normally varies between 0.7 to 1.0 PSF. An option is provided for determining the weight of the air induction system. If k_{AIP} or k_{AIA} is greater than 5.0 the program automatically assumes the value is the weight of the air induction system in pounds.

The k_{NS} term on the input sheet is a nacelle strut factor used when an engine is suspended from an aircraft employing a wing. Enter a unit weight value in PSF in the k_{NS} box. The nominal value is 0.

Legend

W_{EM} = weight of engine mounts - lb

N_E = number of primary engines

W_E = weight of each primary engine - lb

N_{CLF} = aircraft crash load factor

W_{NAC} = weight of each primary engine nacelle - lb

S_{NAC} = wetted area of each nacelle - PSF

PSF = pounds per square foot

W_{AIS} = weight of air induction system - lb

E_{DIA} = primary engine DIA - lb

L_{AID} = length of air inlet duct

The weight of the primary and/or auxiliary engines is determined as part of the engine sizing routine considered elsewhere in the program. There is no provision for determining the engine weight(s) on the weight input sheet. The Δ_{wp} and K_{18} and K_{19} blocks of the input sheet provide a method for adding weight to the engine(s) if desired. (Refer to Table 4-12.

The engine installation weights represent the total weight of items 14, 15, 16, 17, and 19 shown on the weight summary form, Table 4-9. The weights of engine (primary and auxiliary) installation items will vary depending on the type and power-plant arrangement of the configuration being sized. No attempt is made here to describe all the various approaches that may be used to evaluate their weights of the dry engines is provided. Table 4-12 presents the engine installation weights as a percentage of the engine weight for a group of existing helicopters. This may be used as a guide for selecting the weight fraction to be placed in k_{PEI} and/or k_{AEI} on the weight input sheet. An option is provided for determining the weight of the engine installation. If k_{PEI} or k_{AEI} is greater than 1.0, the program automatically assumes the value is the weight of the engine installation in pounds.

Fuel System

The weight of the fuel system, defined as k_{FS} in the propulsion block of the weight input sheet, will vary depending on the capacity, type, and complexity of the system required. For aircraft having simple fuel systems located in the fuselage, sponsons or wing, the value for k_{FS} would range between 0.02 and 0.07; for aircraft requiring self-sealing tanks with more complex systems, the value would range between 0.10 and 0.15. The fuel system factors represent fuel system weight per pound of mission fuel required.

Drive System (Primary and Auxiliary)

The weight of the drive system (primary and auxiliary) including gear boxes, accessory drives, shafting, oil, supports, etc., is derived from the following equation:

$$W_{DS} = 250 a (k_D)^{0.67},$$

Where

$$k_D = \left[\frac{P_X}{N_R} \right] \left[z \right]^{1/4} \left[K_t \right]$$

TABLE 4-12. ENGINE INSTALLATION WEIGHTS

AIRCRAFT	AIRCRAFT ENG. WEIGHT (LB)	ENG. INSTAL. WEIGHT (LB)	PERCENT OF ENGINE WEIGHT
OH-6A	142	36	.254
XH-51A	244	97	.398
BO-105	424	81	.191
UH-1B	474	148	.312
UH-1D	501	147	.293
UH-1N	727	164	.226
UH-34D	1387	260	.187
AH-56A	695	337	.485
CH-46A	600	161	.268
CH-3C	611	130	.213
CH-46D	678	187	.276
CH-46E	886	207	.234
HH-3B	649	136	.210
CH-47A	1160	173	.149
CH-47B	1188	175	.147
CH-47C	1350	244	.181
CH-53A	1432	283	.198
CH-54A	1804	193	.107
CH-54B	2094	394	.188

Note: Engine installation weights include the total weight of the following items:

- Engine Exhaust System
- Engine Cooling
- Engine Controls
- Engine Starting
- Engine Lubrication

Legend

- W_{DS} = weight of the drive system - lb (excluding tail rotor boxes)
- P_X = drive system horsepower rating (tandem rotor
 $P_X = 1.2 \times \text{takeoff rating}$)
- N_R = rotor rpm at takeoff
- Z = number of stages in main rotor drive
- K_t = configuration factor; 1.00 for single rotor,
1.30 for tandem
- a = drive system correcting factor

The drive system adjusting factor a is used to account for type, number of boxes, special features, etc., included in the drive system. Figure 4-35 gives typical examples of the a factor. To determine the k_{PDS} and/or the k_{ADS} figure to place on the weight input sheet, multiply the 250 constant by your selection of a . The k_{PDSZ} and/or k_{ADSZ} (number of stages) must also be placed in their respective locations on the input sheet. As a guide for determining the number of stages to input the following is offered:

	<u>Stages</u>
● Lightweight helicopters (less than 10,000 pounds gross weight)	2
● Medium weight helicopters (10,000 pounds to 30,000 pounds gross weight)	3-4
● Heavy weight helicopters (more than 30,000 pounds gross weight)	4-5

An additional guide in determining the number of stages is to assume one stage for each gear reduction in the drive system. This would include angle boxes. (Assume $\frac{1}{2}$ of a stage for 1:1 angle boxes.) The total additive sum of the stages resulting from this approach would then be placed in their respective k locations on the input sheet.

Tail Rotor Drive System

The weight of the tail rotor drive system, including shafting, etc., is derived from the following equation:

$$W_{DS} = 300 a (k)^{0.8},$$

Where

$$k = \frac{HP_{Total} \times 1.1}{RPM_{Rotor}}$$

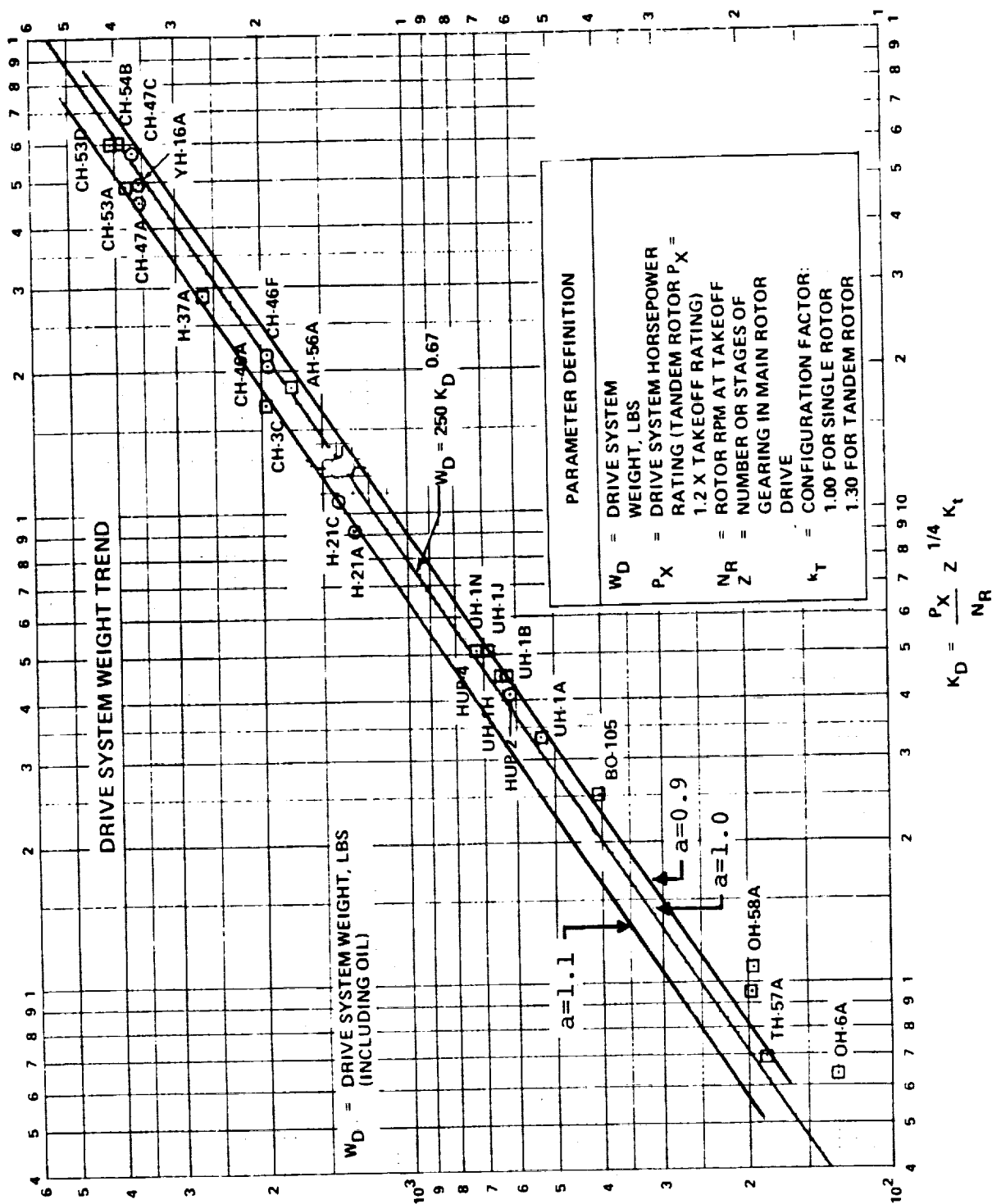


Figure 4-35. Drive System Weight Trend-Primary and Auxiliary.

Legend

W_{DS} = weight of drive system - lb

HP_{Total} = total aircraft horsepower

RPM_{Rotor} = rotor design rpm

a = drive system adjustment factor

The factor a is an adjustment factor used to account for the type, number of boxes, and special features, etc. included in the drive system. Figure 4-36 gives typical examples of the a factor. To determine the k_{DS} value to place on the weight input sheet, multiply the 300 constant by your selection of a .

Flight Controls

The weight of the flight control system will vary depending on the type and system required (manual, power assisted, redundant, dual redundant, etc.) and the type of helicopter being configured (pure, winged, compound, propulsive tail, etc.). Aircraft control systems requiring power assistance and dual or triple redundant components will weigh more than configurations having simple, non-redundant systems. Considerations must be given to these factors when determining the flight control constants to insert on the weight input sheet.

An equation which includes a combined series of weight trend expressions applicable to most any type of helicopter configuration is presented below. It includes factors which can be isolated and applied to the particular vehicle being analyzed. Values for the various k factors described must be put in their proper locations on the weight input sheet. A description of the items comprising each of the control sub-groups is included along with a range of k input values. Refer to the referenced trend curves included for each of the major control groups as an aid in selecting the respective k values.

$$\begin{aligned} W_{FC} = & k_{CC} \left[\frac{W_g}{1000} \right]^{0.41} + k_{RC} \left[\frac{C \sqrt{R W_B}}{1000} \right]^{1.11} + k_{SC} \left[\frac{W_r}{100} \right]^{0.84} \\ & + k_{FW} [W_g] + k_{TM} [W_g] + k_{SAS} + K_{RCA} [W_R] \\ & + k_{SCA} \left[\frac{W_r}{100} \right]^{0.84} + k_{Misc.} \end{aligned}$$

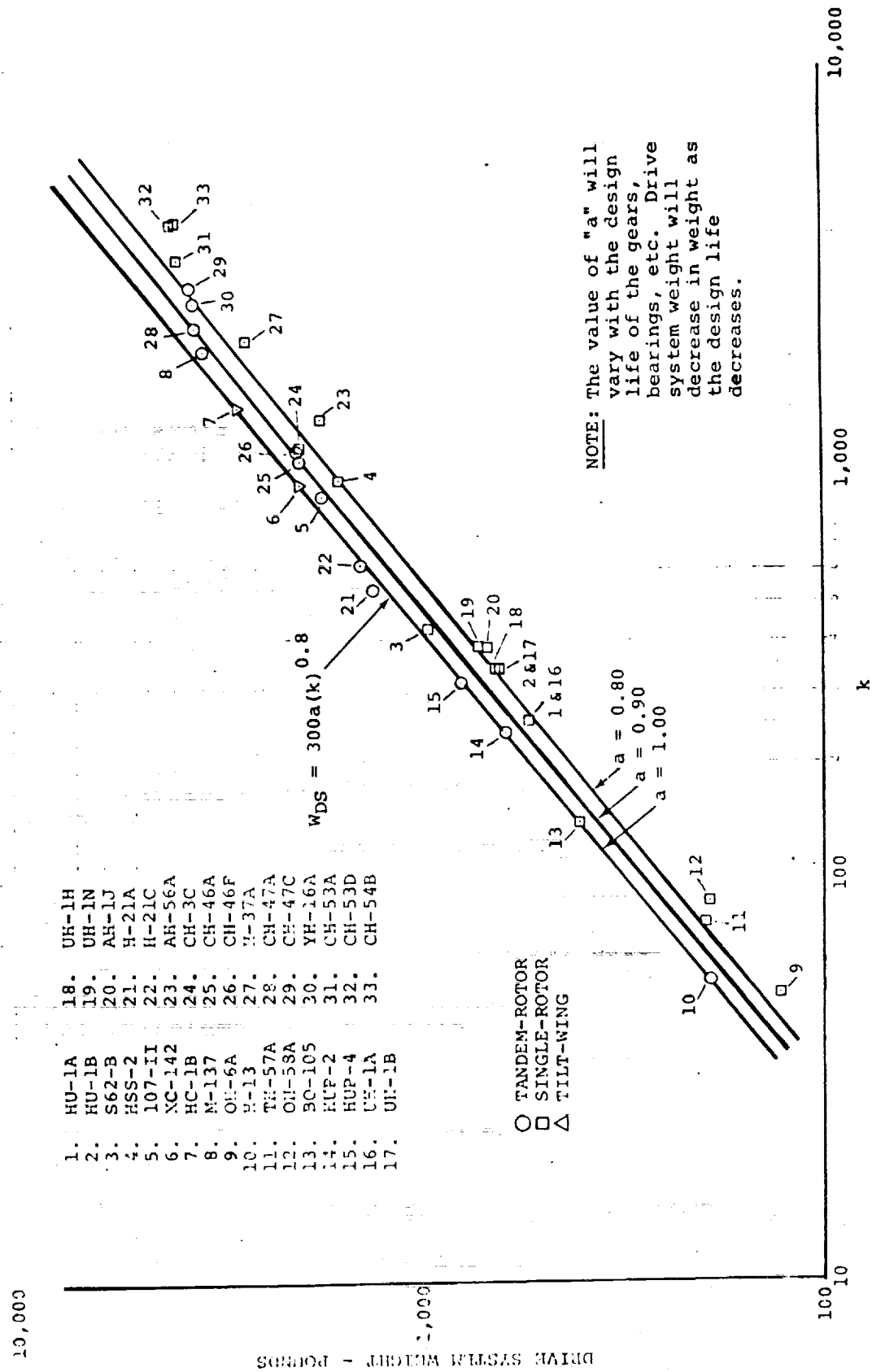


Figure 4-36. Drive System Weight Trend-Tail Rotor.

Legend

W_{FC} = weight of flight controls - lb

W_g = design gross weight - lb

C = rotor blade chord - ft

R = rotor radius - ft

W_b = rotor blade weight per rotor - lb

k_{CC} = constant for cockpit controls = 26

Cyclic and collective control sticks and linkages, pedals, cables and rods (Figure 4-37)

k_{RC} = constant for main rotor controls = 18 to 23

All components from and including the power actuators up through the pitch links. Major items included are the actuators, swashplate, and pitch links (Figure 4-38)

k_{SC} = constant for main rotor systems and hydraulics = 25 to 35

All components between the cockpit controls and the rotor controls including actuators, artificial feel system, mechanical programmer, bellcranks, rods, idlers, etc. (Figure 4-39)

Main hydraulic systems including pumps, reservoirs, accumulators, filters, valves, lines, fluid, and supports (Figure 4-39)

k_{FW} = constant for conventional fixed-wing controls = 0.005 to 0.020, depending on complexity and number of functions required

All components, actuators, and supports associated with moving the control surfaces - LE umbrellas, flaperons, spoilers, and tail surfaces

k_{TM} = constant for tilting mechanism - 0.005 to 0.015

All components and supports required to tilt the wing including actuators, power control units, mechanical system, fittings, and hardware. The k_{TM} value will vary proportionately with the hinge moment and/or wing transition rate required.

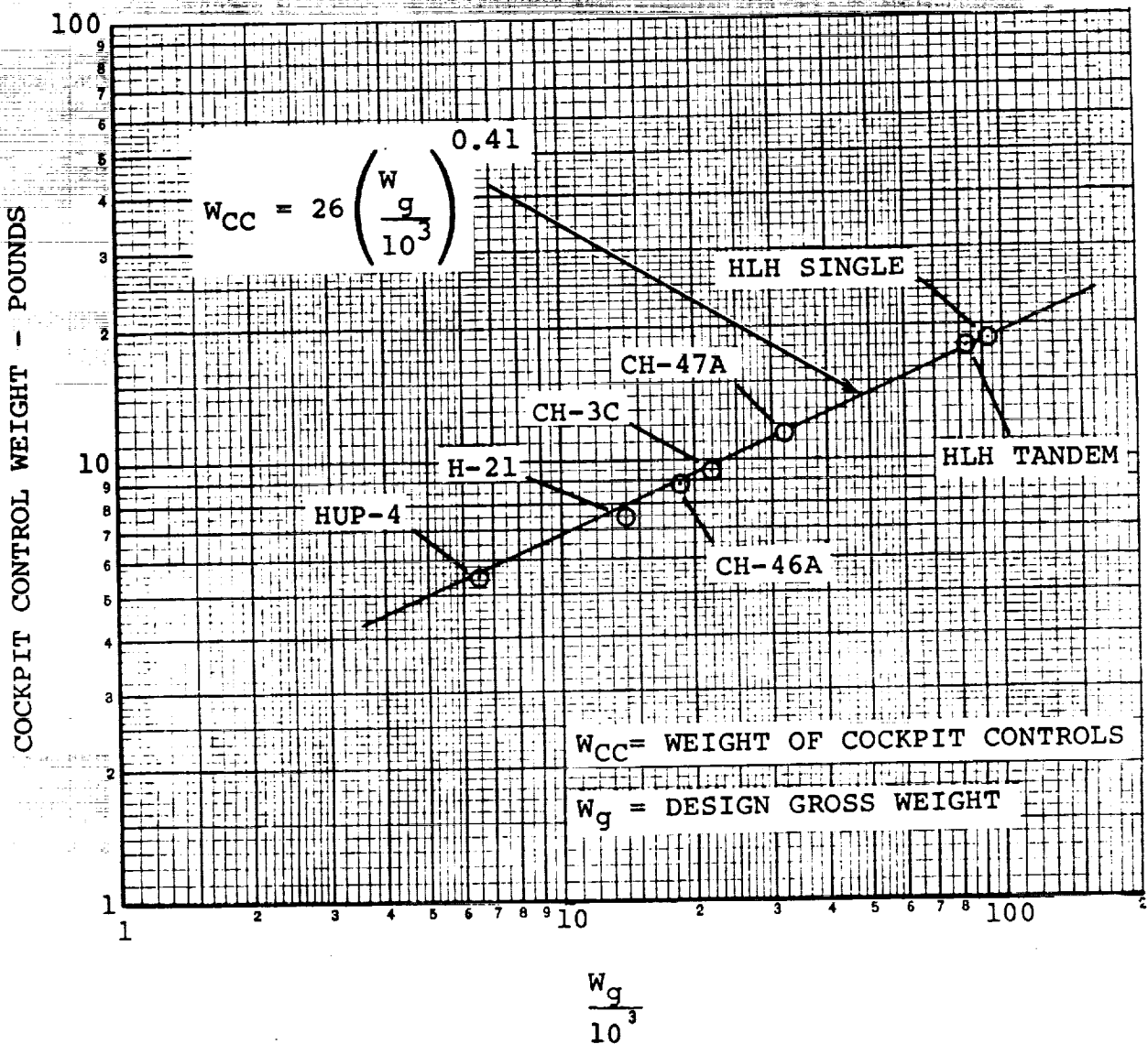


Figure 4-37. Cockpit Controls Weight Trend.

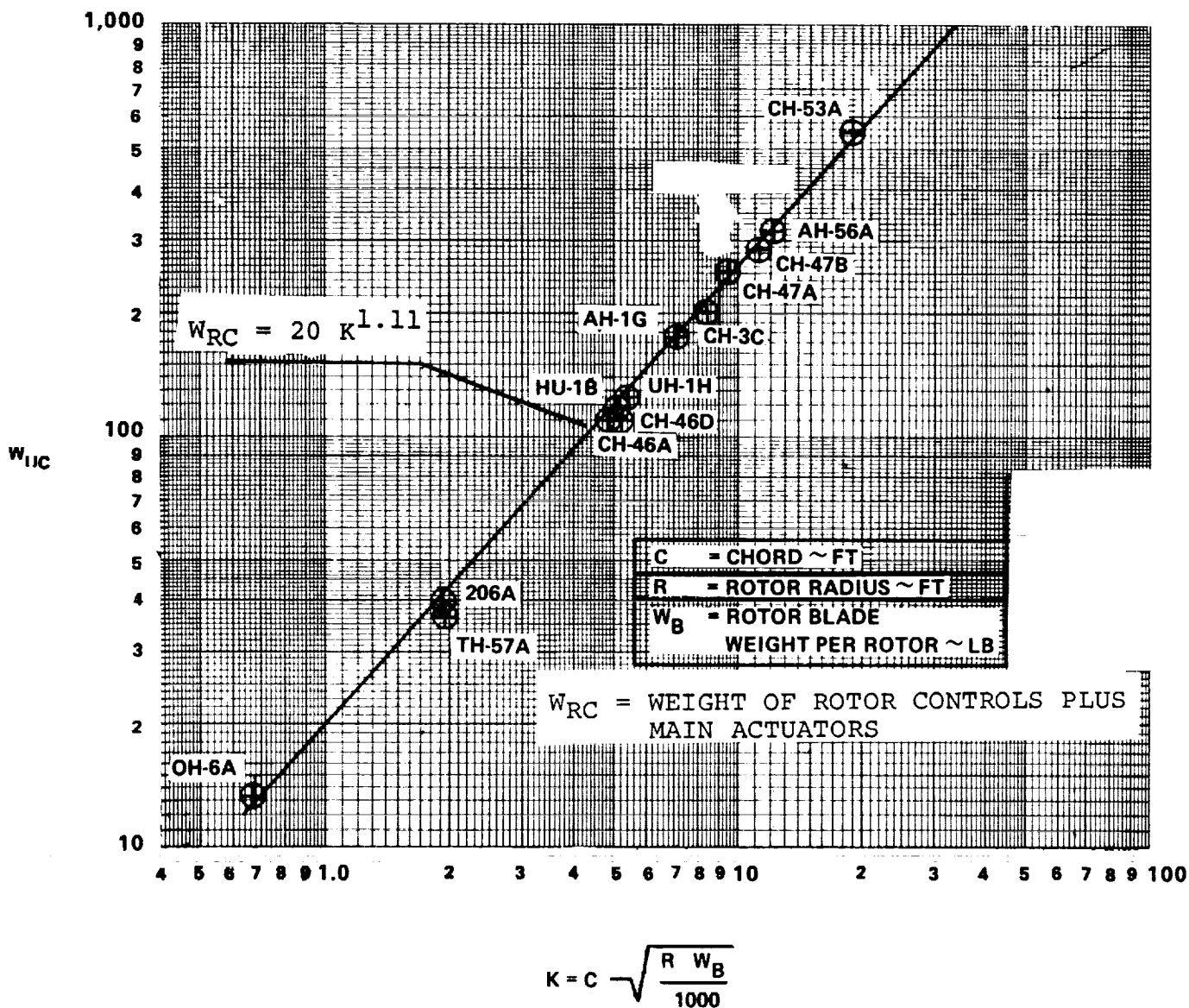


Figure 4-38. Rotor Controls Weight Trend.

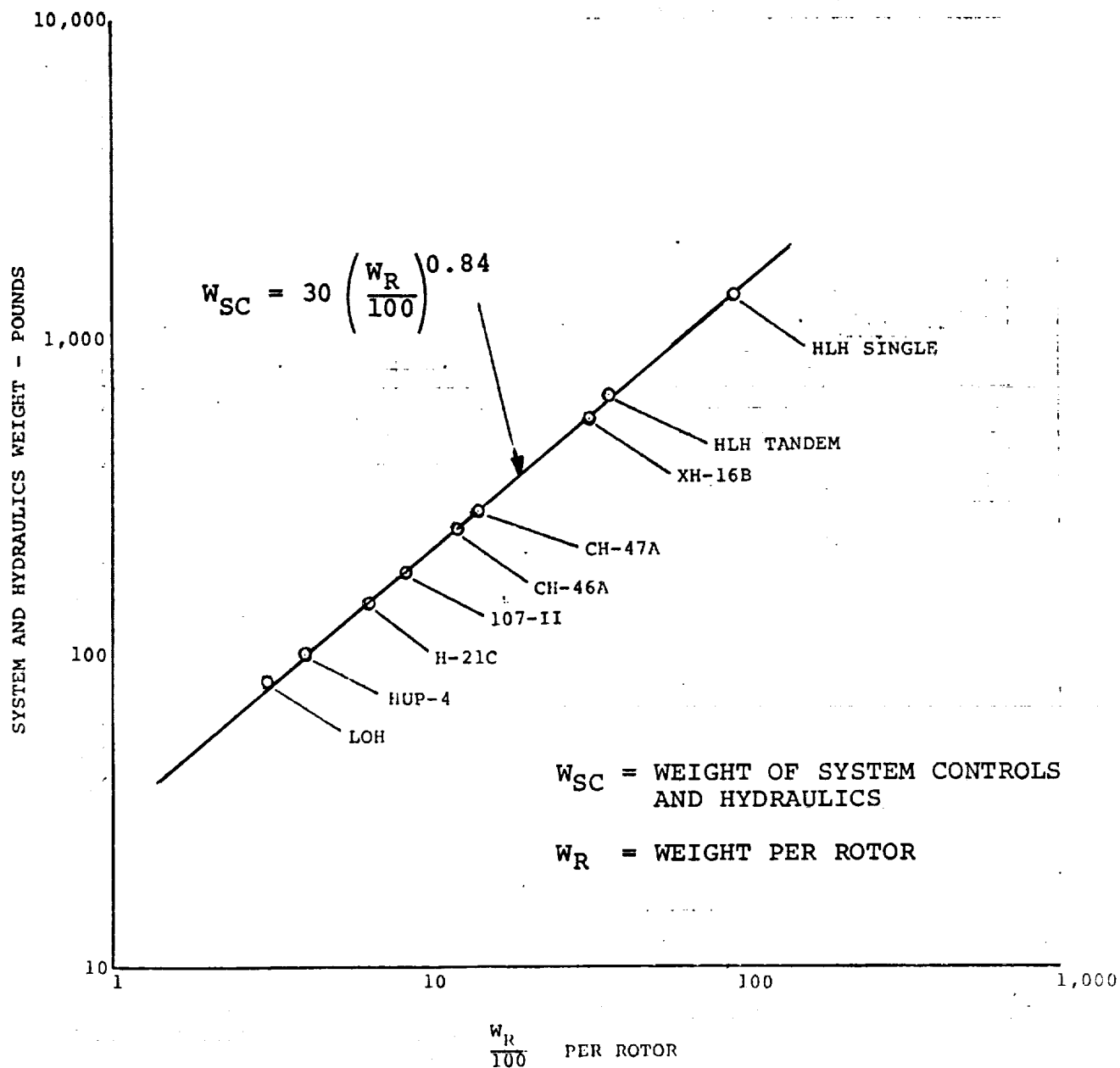


Figure 4-39. Rotor System and Hydraulics Weight Trend.

k_{SAS} = constant for stability-augmented system =
20 pounds to 100 pounds, depending on system
required

k_{RCA} = constant for auxiliary rotor controls

Similar to k_{RC} - provides rotor control weights
for auxiliary propulsive systems (pusher props,
ducted fan, etc., Figure 4-40)

k_{SCA} = constant for auxiliary rotor system controls

Similar to k_{SC} - provides rotor system control
weights for auxiliary propulsive systems (pusher
props, ducted fans, etc., Figure 4-39)

$k_{Misc.}$ = estimated weight input in pounds for any items
not covered above

Fixed Equipment

The weight of the fixed equipment is included in the weight empty and consists of the following groups: auxiliary power-plant, instruments, hydraulics and pneumatics, electrical, avionics, armament, furnishings and equipment, air-conditioning, anti-icing and load and handling (Table 4-9).

The weight of the fixed equipment will vary with the type and requirements of the aircraft under study. The largest variation in fixed equipment weights usually appears in the avionics and the furnishings and equipment groups. The avionics group reflects communication and navigational requirements; the furnishings and equipment group normally reflects cabin size and personnel accommodations (pilots seats, troop seats, etc.). Table 4-13 presents some typical examples of the fixed equipment weights for some existing military helicopters.

The total weight of the fixed equipment, W_{FE} , must be placed in the W_{FE} block of the weight input sheet.

Fixed Useful Load

The weight of the fixed useful load represents a portion of the useful load. It includes the crew, trapped and unusable fuel and oil, guns, weapons, racks or pylons and any other fixed items of useful load which makes the aircraft operational. Typical weights for fixed useful load items are included in Table 4-13 as a guide for inputting a number in the W_{FUL} block of the weight input sheet.

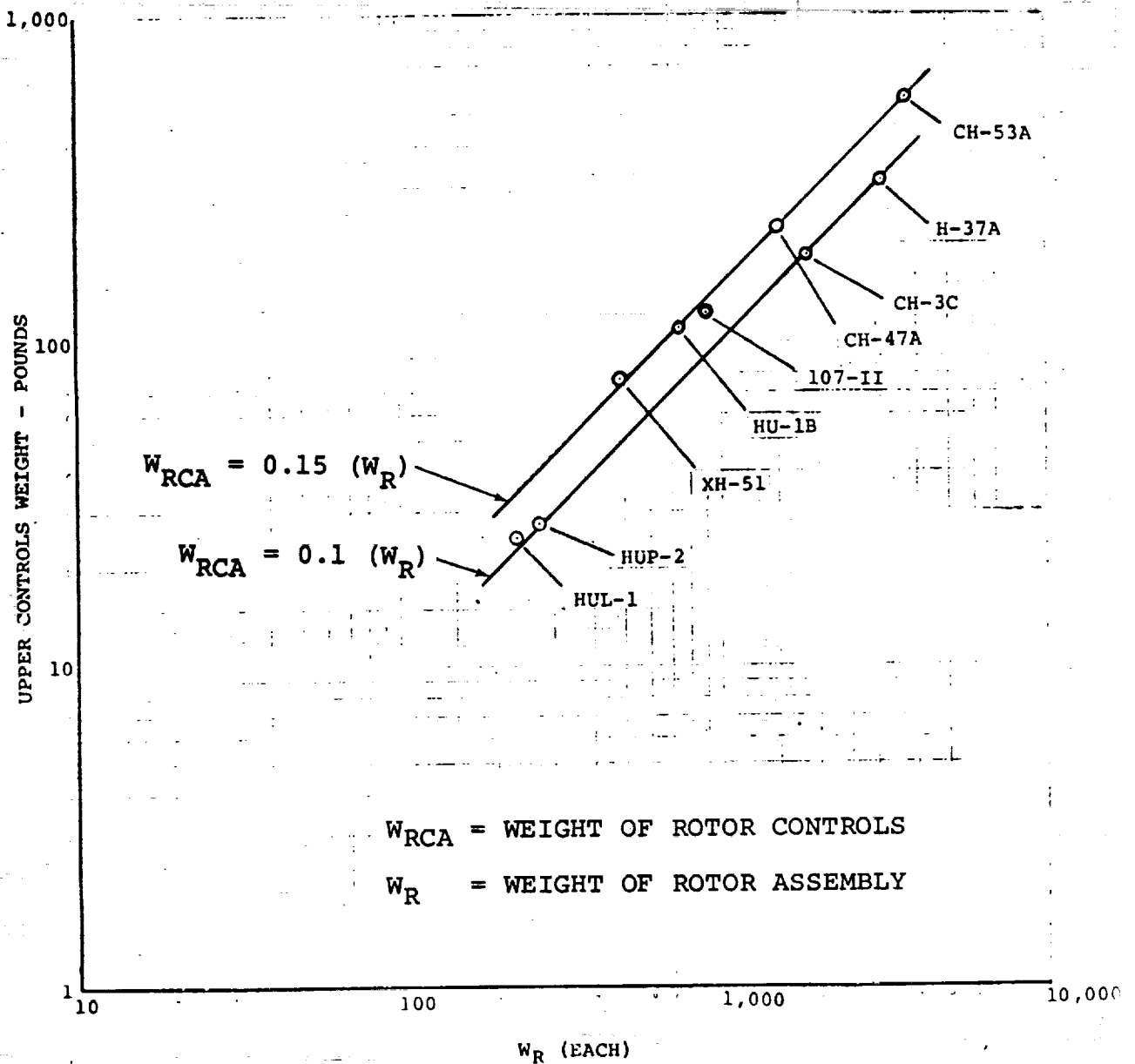


Figure 4-40. Auxiliary Rotor Controls Weight Trend.

TABLE 4-13
FIXED EQUIPMENT AND FIXED USEFUL LOAD WEIGHTS

CH-46A CH-47A CH-53A CH-3C AH-56A 107-II-10

WING						
ROTOR						
TAIL						
SURFACES						
ROTOR						
BODY						
BASIC						
SECONDARY						
ALIGNING GEAR GROUP						
ENGINE SECTION						
PROPULSION GROUP						
ENGINE INST'L						
EXHAUST SYSTEM						
COOLING						
CONTROLS						
STARTING						
PROPELLER INST'L						
LUBRICATING						
FUEL						
DRIVE						
FLIGHT CONTROLS						
AUX. POWER PLANT	100	103	204	224	136	-
INSTRUMENTS	169	161	393	232	126	98
HYDR. & PNEUMATIC	163	227	112	66	86	-
ELECTRICAL GROUP	620	560	594	450	377	583
AVIONICS GROUP	386	274	612	437	609	602
ARMAMENT GROUP	-	-	18	-	548	-
FURN. & EQUIP. GROUP	788	896	971	569	273	1300
ACCOM. FOR PERSON.	182	391	298	208	165	555
MISC. EQUIPMENT	77	107	384	87	73	601
FURNISHINGS	481	329	214	175	-	-
EMERG. EQUIPMENT	48	69	75	99	35	144
AIR CONDITIONING	128	145	237	123	75	128
ANTI-ICING GROUP	186	34	77	37	42	13
LOAD AND HANDLING GP.	305	260	358	189	-	14
FIXED EQUIP. WEIGHT	2845	2660	3576	2327	2272	2738
CREW	540	600	660	645	400	400
TRAPPED LIQUIDS	20	41	18	20	31	28
ENGINE OIL	30	28	48	33	32	31
SURVIVAL KIT				63	-	
ARMOR					399	
GUNS					1235	
ATTENDANT						150
BAGGAGE						370
FUEL						
FIXED USEFUL LOAD	590	669	726	761	2097	979

Payload

The weight of the payload is determined by the mission requirements. The total weight of the payload must be put in the W_{PL} block of the weight input sheet.

Incremental Group Weights

The incremental group weights section of the weight input sheet is provided to enable the user to add fixed increments of weight where desired. Definitions and values for some of the items in this group have already been discussed. ΔW_{FC} , ΔW_p , and ΔW_{ST} represent incremental weights of the flight controls group, propulsion group, and structural group, respectively. Any value inserted in the incremental group weight section remains constant regardless of gross weight. The nominal value for any block in this section is 0, except for R_M which is nominally 1.0.

Group Weight Information

The nominal value for items in this section of the weight input sheet is 0, except as noted in the text. All blocks must be filled in. Definitions and constants for the various k factors have been previously discussed in the respective subgroup definitions.

Multiplicative Factors

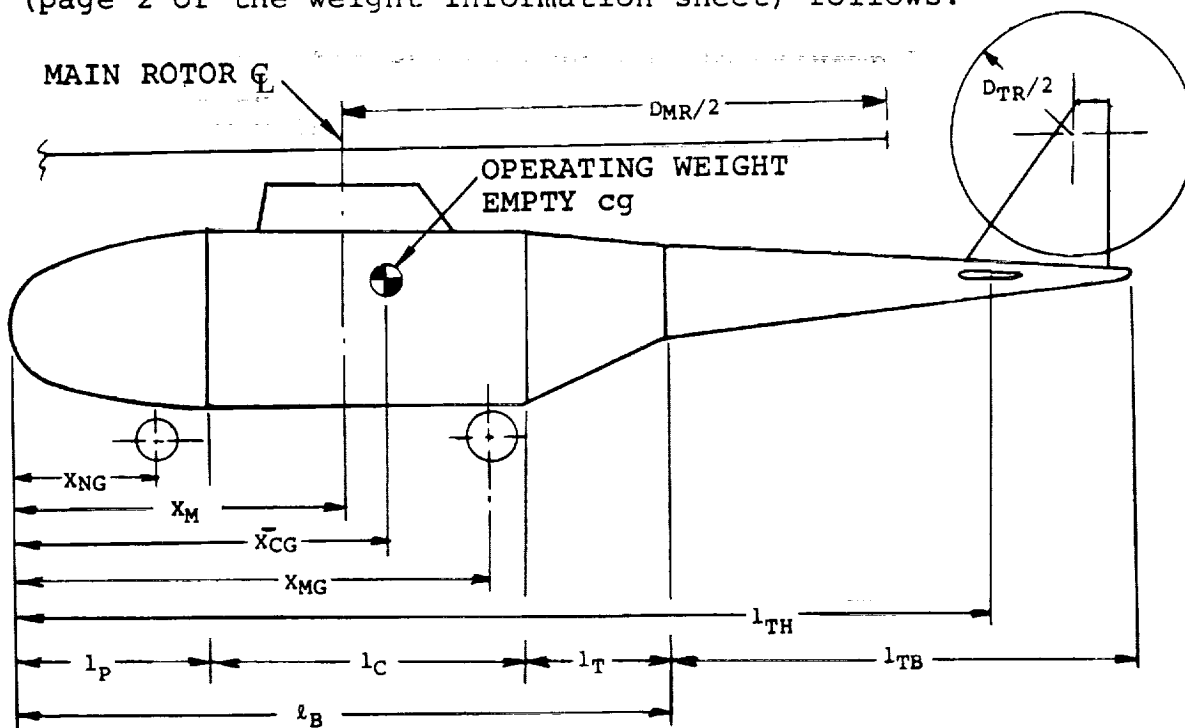
The multiplicative factors described as K_1 through K_{20} on the weight input sheet provide the capability of performing weight sensitivity studies. The factors are nominally 1. All blocks must be filled in. To vary the weight of any subgroup (k_{CC} , k_B , k_{DS} , etc.), insert the desired value in the appropriate multiplicative box. Refer to Table 4-14 to relate the various k factors with their respective groups. Inserting a value of 1.1 would increase the weight of the respective group by 10 percent; a value of 0.9 would decrease it by 10 percent, etc. The values in this group will vary with gross weight.

TABLE 4-14
MULTIPLICATIVE FACTORS

K	LOCATION	LETTER CODE	DESCRIPTION
K1	2654	W _{RC}	WEIGHT OF MAIN ROTOR CONTROLS
K2	2655	W _{SC}	WEIGHT OF MAIN ROTOR SYSTEM CONTROLS
K3	2656	W _{FW}	WEIGHT OF FIXED WING CONTROLS
K4	2657	W _{RCA}	WEIGHT OF AUXILIARY ROTOR CONTROLS
K5	2658	W _{SCA}	WEIGHT OF AUXILIARY ROTOR SYSTEM CONTROLS
K6	2659	W _B	WEIGHT OF BODY
K7	2660	W _{LG}	WEIGHT OF LANDING GEAR
K8	2661	W _W	WEIGHT OF WING
K9	2662	W _{HT}	WEIGHT OF HORIZONTAL TAIL
K10	2663	W _{NAC}	WEIGHT OF PRIMARY NACELLE
K11	2664	W _{NACA}	WEIGHT OF AUXILIARY NACELLE
K12	2665	W _{PRB}	WEIGHT OF PRIMARY ROTOR BLADES
K13	2666	W _{PH}	WEIGHT OF PRIMARY ROTOR HUB
K14	2667	W _{TR}	WEIGHT OF TAIL ROTOR
K15	2668	W _{AR}	WEIGHT OF AUXILIARY ROTOR
K16	2669	W _{PDS}	WEIGHT OF PRIMARY DRIVE SYSTEM
K17	2670	W _{ADS}	WEIGHT OF AUXILIARY DRIVE SYSTEM
K18	2671	W _{PE}	WEIGHT OF PRIMARY ENGINE
K19	2672	W _{AE}	WEIGHT OF AUXILIARY ENGINE
K20	2673	W _{TRDS}	WEIGHT OF TAIL ROTOR DRIVE SYSTEM

4.11.2 Aircraft Balance

A preliminary aircraft balance for single rotor helicopters is included in the program which locates the main rotor system $\bar{c}$ relative to the required center of gravity of the operating weight empty of the aircraft. A description of the input values to be included on the weight-balance information sheet (page 2 of the weight information sheet) follows:



- Loc 2678 Center of gravity of the fuselage, measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2679 Distance in feet between the OWE center of gravity and the main rotor center line (negative value (-) is forward of OWE cg, plus value (+) is aft of OWE cg).
- Loc 2680 Center of gravity of the nose gear measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2681 Center of gravity of the main gear, measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2682 Center of gravity of the primary engine package, measured from the nose of the aircraft as a fractional part of l_B . The engine package consists of the engine section, engine, engine installation and fuel system.

- Loc 2683 Center of gravity of the primary drive system measured as a fractional part of the distance between the main rotor and tail rotor. The tail rotor drive system weight and balance are computed and located automatically.
- Loc 2684 Center of gravity of the avionics system, measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2685 Center of gravity of the furnishings and equipment, cockpit controls and a portion of the useful load (pilot and copilot), measured from the nose of the aircraft as a fractional part of l_B .
- Loc 2686 Center of gravity of the APU, measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2687 Center of gravity of the auxiliary engine package, measured from the nose of the aircraft, as a fractional part of l_B . The auxiliary engine package consists of the auxiliary engine section, auxiliary engine and auxiliary engine installation.
- Loc 2688 Center of gravity of the auxiliary drive system as a fractional part of the distance between the auxiliary engine and auxiliary rotor system.
- Loc 2689 Center of gravity of the auxiliary rotor and auxiliary rotor controls, measured from the nose of the aircraft, as a fractional part of l_{TB} .
- Loc 2690 Center of gravity of the system controls, measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2691 Center of gravity of the auxiliary system controls, measured from the nose of the aircraft, as a fractional part of l_B .
- Loc 2692 Total weight of the avionics group weight plus SAS input from flight controls column of weight input sheet.
- Loc 2693 Total weight of furnishings and equipment group located in the pilot's compartment and the weight of the cockpit controls (CC) as determined from the input constant in flight controls column of weight input sheet.
- Loc 2694 Total weight of auxiliary power unit (APU) installation.

Loc 2695 Fractional part of fixed useful load (pilot, co-pilot, etc.) located in the pilot's compartment.

Loc 2696 Tail boom weight expressed as a fractional part of the computed body weight. The center of gravity of the tail boom is automatically computed in the program.

Items of the operating weight empty not included in the location descriptions presented above are located at the aircraft center of gravity as computed by the balance subroutine. An example of a completed weight-balance information sheet for a typical single rotor helicopter is shown below.

WEIGHT-BALANCE INFORMATION
(Required Only When MRPIND > 0)

Variable	Loc.	Value
(X_{CGF}/l_B)	2678	0.425
ΔCG_R	2679	0.10
(X_{NG}/l_B)	2680	0.216
(X_{MG}/l_B)	2681	0.754
(X_{PE}/l_B)	2682	0.727
$(X_{PDS}/\Delta X_{PDS})$	2683	0.593
(X_{AV}/l_B)	2684	0.485

Variable	Loc.	Value
(X_{FURN}/l_B)	2685	0.334
(X_{APU}/l_B)	2686	0.424
(X_{AE}/l_B)	2687	0
(X_{ADS}/X_{ADS})	2688	0
(X_{AR}/l_{TB})	2689	0
(X_{SC}/l_B)	2690	0.581
(X_{ASC}/l_B)	2691	0

Variable	Loc.	Value
W_{AV}	2692	469
W_{FURN}	2693	416
W_{APU}	2694	172
K_{FULS}	2695	0.731
K_{TBBS}	2696	0.130

Name _____
Date _____Page _____
Model _____
Report _____

GROUP WEIGHT STATEMENT

AIRCRAFT

(INCLUDING ROTORCRAFT)

ESTIMATED - CALCULATED - ACTUAL

(Cross Out Those Not Applicable)

CONTRACT NO. _____

AIRCRAFT, GOVERNMENT NO. _____

AIRCRAFT, CONTRACTOR NO. _____

MANUFACTURED BY _____

ENGINE		MAIN	AUX
	MANUFACTURED BY		
	MODEL		
	NO.		
	TYPE		

PAGES REMOVED	PAGE NO.

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GROUP WEIGHT STATEMENT
WEIGHT EMPTY

Name _____

Page _____

Date _____

Model _____

Report _____

1	WING GROUP				
2	BASIC STRUCTURE - CENTER SECTION				
3	INTERMEDIATE PANEL				
4	OUTER PANEL				
5	GLOVE				
6	SECONDARY STRUCTURE (Incl. Wing Fold Weight Lbs.)				
7	AILERONS (Incl. Balance Weight Lbs.)				
8	FLAPS - TRAILING EDGE				
9	LEADING EDGE				
10	SLATS				
11	SPOILERS				
12					
13					
14	ROTOR GROUP				
15	BLADE ASSEMBLY				
16	HUB & HINGE (Incl. Blade Fold Weight Lbs.)				
17					
18					
19	TAIL GROUP				
20	BASIC & SECONDARY STRUCT. - STABILIZER				
21	FIN (Incl. Dorsal)				
22	VENTRAL				
23	ELEVATOR (Incl. Balance Weight Lbs.)				
24	RUDDERS (Incl. Balance Weight Lbs.)				
25	TAIL ROTOR - BLADES				
26	HUB & HINGE				
27					
28	BODY GROUP				
29	BASIC STRUCTURE - FUSELAGE or HULL				
30	BOOMS				
31	SECONDARY STRUCTURE FUSELAGE or HULL				
32	BOOMS				
33	SPEEDBRAKES				
34	DOORS, RAMPS, PANELS, & MISC.				
35					
36					
37	ALIGNING GEAR GROUP (Type:)				
38	LOCATION	Running Gear*	Arrest Gear*	Structure	Controls
39					
40					
41					
42					
43					
44					
45	ENGINE SECTION or NACELLE GROUP				
46	BODY INTERNAL				
47	EXTERNAL				
48	WING INBOARD				
49	OUTBOARD				
50					
51					
52	AIR INDUCTION SYSTEM				
53	DOORS, PANELS, & MISC.				
54					
55					
56					
57	TOTAL STRUCTURE (To Be Brought Forward)				

*Change to Floats & Struts for Water Type Gear

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GROUP WEIGHT STATEMENT
WEIGHT EMPTY

Page _____

Name _____

Model _____

Date _____

Report _____

		Auxiliary	Main
1	PROPULSION GROUP		
2	ENGINE INSTALLATION		
3			
4	ACCESSORY GEAR BOXES & DRIVE		
5			
6	EXHAUST SYSTEM		
7	ENGINE COOLING		
8	WATER INJECTION		
9	ENGINE CONTROL		
10	STARTING SYSTEM		
11	PROPELLER INSTALLATION		
12	SMOKE ABATEMENT		
13	LUBRICATING SYSTEM		
14	FUEL SYSTEM		
15	TANKS . PROTECTED		
16	UNPROTECTED		
17	PLUMBING, etc		
18	DRIVE SYSTEM		
19	GEAR BOXES, LUB SY & ROTOR BRK		
20	TRANSMISSION DRIVE		
21	ROTOR SHAFTS		
22	JET DRIVE		
23			
24	FLIGHT CONTROLS GROUP		
25	COCKPIT CONTROLS (Autopilot Lbs.)		
26	SYSTEMS CONTROLS		
27			
28			
29	AUXILIARY POWER PLANT GROUP		
30	INSTRUMENTS GROUP		
31	HYDRAULIC & PNEUMATIC GROUP		
32			
33	ELECTRICAL GROUP		
34			
35	AVIONICS GROUP		
36	EQUIPMENT		
37	INSTALLATION		
38			
39	ARMAMENT GROUP (Incl. Passive Prot. Lbs.)		
40	FURNISHINGS & EQUIPMENT GROUP		
41	ACCOMMODATION FOR PERSONNEL		
42	MISCELLANEOUS EQUIPMENT		
43	FURNISHINGS		
44	EMERGENCY EQUIPMENT		
45			
46	AIR CONDITIONING GROUP		
47	ANTI ICING GROUP		
48			
49	PHOTOGRAPHIC GROUP		
50			
51	LOAD & HANDLING GROUP		
52	AIRCRAFT HANDLING		
53	LOAD HANDLING		
54			
55	MANUFACTURING VARIATION		
56	TOTAL FROM PAGE 2		
57	WEIGHT EMPTY		

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GROUP WEIGHT STATEMENT
USEFUL LOAD AND GROSS WEIGHT

Page _____

Model _____

Name _____

Report _____

Date _____

1	LOAD CONDITION								
2									
3	CREW (No.)								
4	PASSENGERS (No.)								
5	FUEL	Location	Type	Gals.					
6	UNUSABLE								
7	INTERNAL								
8									
9									
10									
11	EXTERNAL								
12									
13									
14	OIL								
15	TRAPPED								
16	ENGINE								
17									
18	FUEL TANKS (Location)								
19	WATER INJECTION FLUID (Gals.)								
20									
21	BAGGAGE								
22	CARGO								
23									
24	GUN INSTALLATIONS								
25	GUNS	Location	Fix. or Flex	Quantity	Caliber				
26									
27									
28	AMMO								
29									
30									
31	SUPPTS								
32	WEAPONS INSTALL. (Incl. Submarine Detection Expendables)								
33									
34									
35									
36									
37									
38									
39									
40									
41									
42									
43									
44									
45									
46	EQUIPMENT								
47									
48	SURVIVAL KITS & LIFE RAFTS								
49									
50	OXYGEN								
51									
52									
53									
54									
55	TOTAL USEFUL LOAD								
56	WEIGHT EMPTY								
57	GROSS WEIGHT								

*If Removable and Specified as Useful Load

**List Stores, Missiles, Sonobuoys, etc. Followed by Racks, Launchers, Chutes, etc. Net Part of Weight Empty
List Identification, Location, and Quantity for All Items Shown Including Installation

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GROUP WEIGHT STATEMENT
DIMENSIONAL AND STRUCTURAL DATA

Page _____

Name _____

Model _____

Date _____

Report _____

1	WING, ROTOR & TAIL GROUPS	SPAN OR RADIUS, FT.	SPAN AT **** 25% CHORD	THEO. ROOT ** CHORD IN.	MAX. THICK ** ROOT CHORD IN.	THEO. TIP CHORD IN. **	MAX. THICK ** TIP CHORD IN.
2	WING						
3							
4	MAIN ROTOR (Blades/Rotor)						
5	TAIL ROTOR (Blades/Rotor)						
6	HORIZ. TAIL						
7	VERT. TAIL						
8							
9	AREAS - (Sq. Ft.)	Wing	MAIN ROTOR BLADE AREA	TAIL ROTOR BLADE AREA	Horiz. Tail	Vert. Tail	Dorsal
10	(Theo. for Wing & Rotor, All Others Exposed)						
11		Speed Brks.	Flaps (L.E.)	Flaps (T.E.)	Slats	Spoilers	Ailerons
12	AREAS - (Sq. Ft.)						
13	BODY & NACELLE GROUPS	Length (Ft.)	Depth (Ft.)	Width (Ft.)	WETTED AREA SQ. FT.	Vol. (Cu. Ft.)	HOL PRESS. CU. FT.
14	FUSELAGE or HULL						
15	BOOMS						
16	NACELLES						
17							
18							
19	ALIGHTING GEAR GROUP	Length - Oleo Ext.	Oleo Travel	Length - Arrest Hook			
20		Axle to & Trunnion	Ext. to Collapsed	Hook Trunnion to Pt.			
21	LOCATION						
22	DIMENSIONS (Inches)						
23							
24	PROPULSION GROUP						
25	ENGINES	SLS THRUST IN LBS. ENG. WITH AFTERBURNER	SLS THRUST IN LBS. ENG. WITHOUT AFTERBURNER	MAX. SLS SHAFT H.P.	SHAFT R.P.M. AT MAX. H.P.		
26	MAIN						
27	AUXILIARY						
28	ROTOR DRIVE SYSTEM	Design H.P.	Input R.P.M.	OUTPUT R.P.M. AT ROTOR	INTER ROTOR R.P.M.	NUMBER GEAR BOXES	
29							
30		Protected	Unprotected	Integral			
31	FUEL - INTERNAL *** LOCATION	No. Tanks	Gallons	No. Tanks	Gallons	No. Tanks	Gallons
32	WING						
33	FUSELAGE						
34	EXTERNAL ***						
35							
36	OIL						
37	ELECTRICAL & LOAD & HANDLING GROUPS	QUAN. MAIN GENERATORS	GENERATOR OUTPUT D.C.	GENERATOR OUTPUT A.C.		CARGO FLOOR AREA	
38							
39							
40	STRUCTURAL DATA - CONDITION	BODY PLUS INT. CONTENT - LBS.	EXTERNAL WT. ON BODY	FUEL IN WINGS - LBS.		DESIGN GROSS WEIGHT	Ult. L.F.
41	FLIGHT - MANEUVER						
42	GUST						
43	LANDING						
44							
45	MAX. GROSS WITH ZERO WING FUEL						
46	CATAPULTING						
47	LIMIT LANDING SINK SPEED (Ft./Sec.)						
48	(FT. ASSUMED FOR LANDING DESIGN CONDITION - PS. WT.)						
49	STALL SPD. LDG. CONFIG. POWER OFF						
50	PRESUMED CABIN ALT. DESIGN PRESS. DIFFERENTIAL (FLIGHT P.S.I.)						
51	ROTOR TIP SPD. AT DESIGN LIMIT	R.P.M.	Power	Ft./Sec.	CONTRACTOR DESIGN FACTOR		
52							
53	% DESIGN LOAD	Wing		Rotor		Rotor	
54	DESIGN SPEED AT S.L. (Knots)	Level		Dive			
55	DESIGN SPD. AT OTHER ALTITUDES		Alt.		Alt.		
56							
57	DCPR WEIGHT (Airframe)						

*Nose to aft tip of fuselage (excluding equipment protuberances)

**Parallel to & at & Aircraft for Wing & Tail. Insert inches from & Rotor for Rotors.

***Total Usable Capacity

****Insert inches from & Rotor to Blade Attachment for Rotors

MIL-STD-1374 PART I

Name _____

Date _____

GROUP WEIGHT STATEMENT

Page _____

Model _____

Report _____

AIRFRAME WEIGHT

The Airframe Weight to be entered on line 57 of page 5 of the Group Weight Statement should be derived here in detail showing those items deducted from weight empty as required by the document "Cost Information Reports (CIR) for Aircraft, Missiles, and Space Systems" dated 21 April, 1966, or subsequent revisions thereto. Airframe weight is the same as previously called AMPR and DCPR and is not to be confused with "Work Breakdown Structure (WBS) Airframe Cost Definition."

WEIGHT EMPTY
DEDUCT THE FOLLOWING ITEMS
(ITEMIZE)

AIRFRAME WEIGHT

4.12 PERFORMANCE CALCULATIONS SUBPROGRAM

The flow chart of the control loop for the performance calculations subprogram is shown in Figure 4-41. This routine monitors the flow during calculation of mission performance data and calculates the total fuel required at the end of the mission.

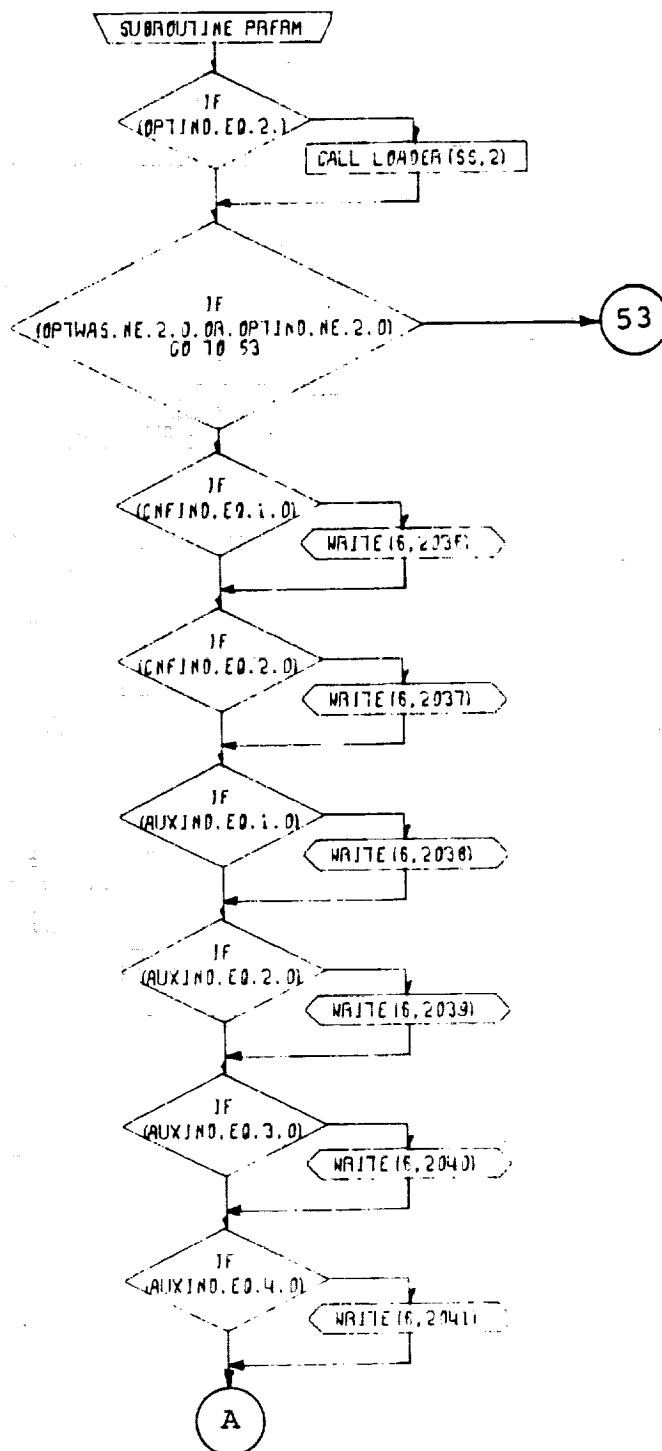


Figure 4-41. Performance Calculations Subprogram Flow Chart
(Part 1 of 5).

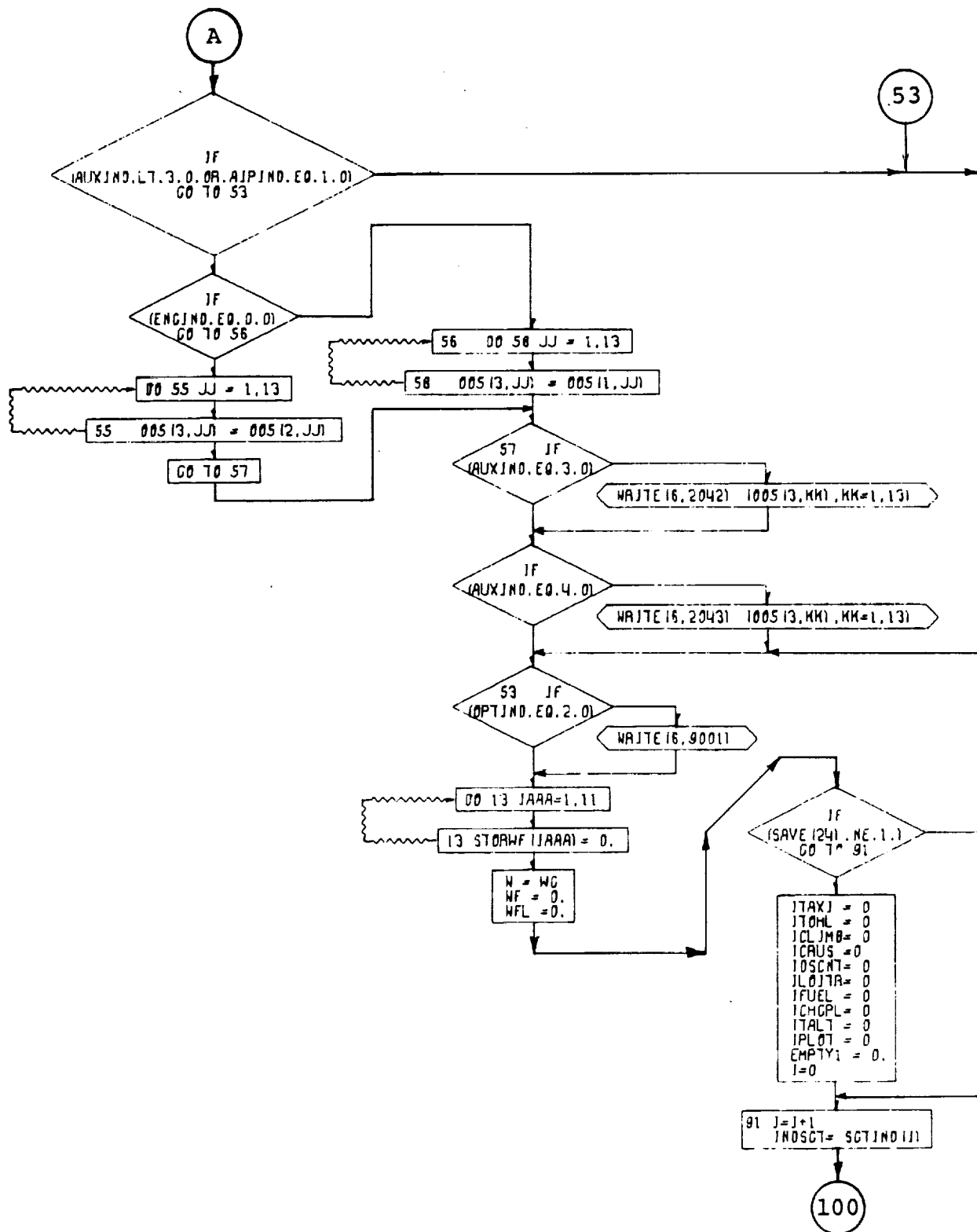


Figure 4-41. Performance Calculations Subprogram Flow Chart (Part 2 of 5).

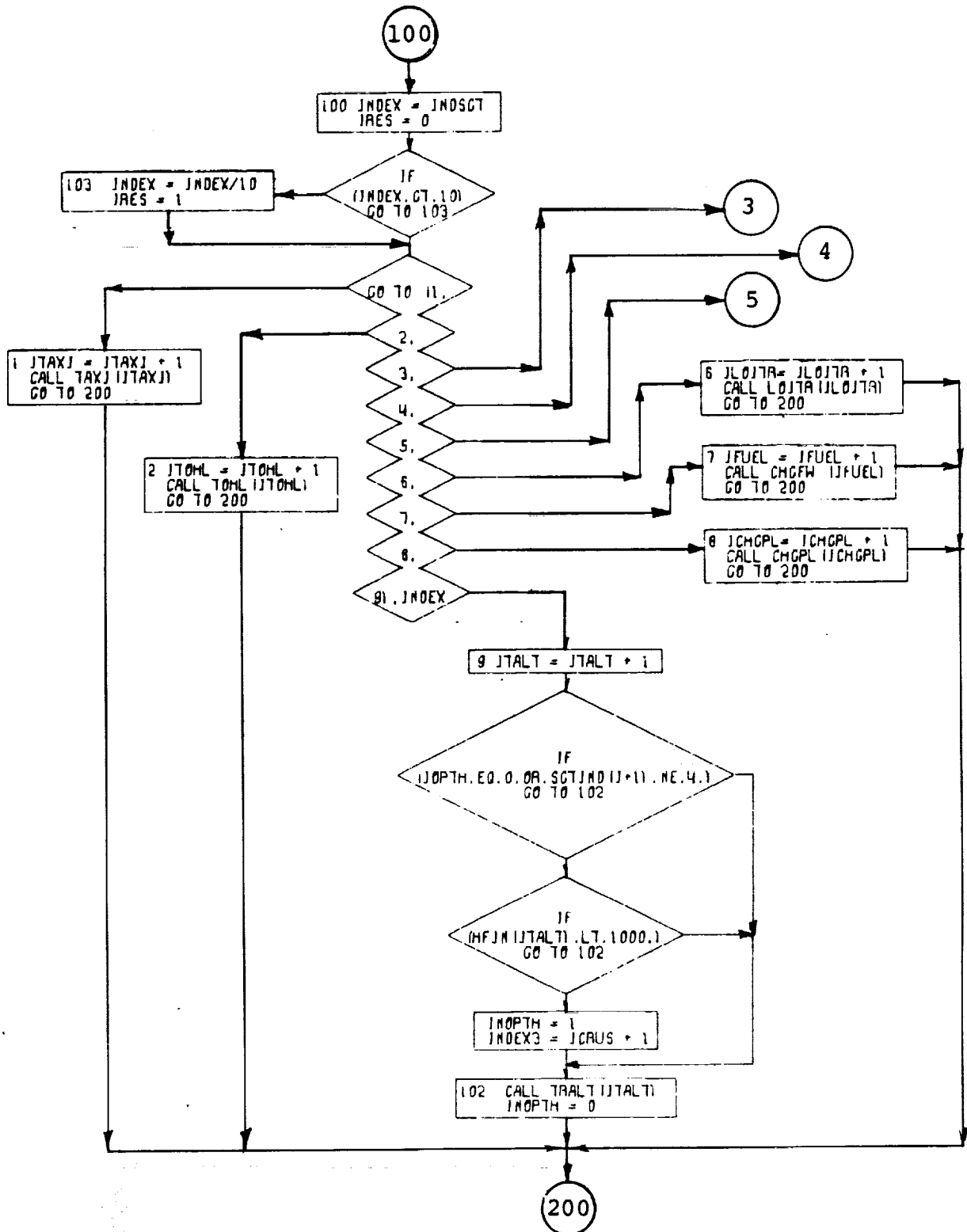


Figure 4-41. Performance Calculations Subprogram Flow Chart (Part 3 of 5).

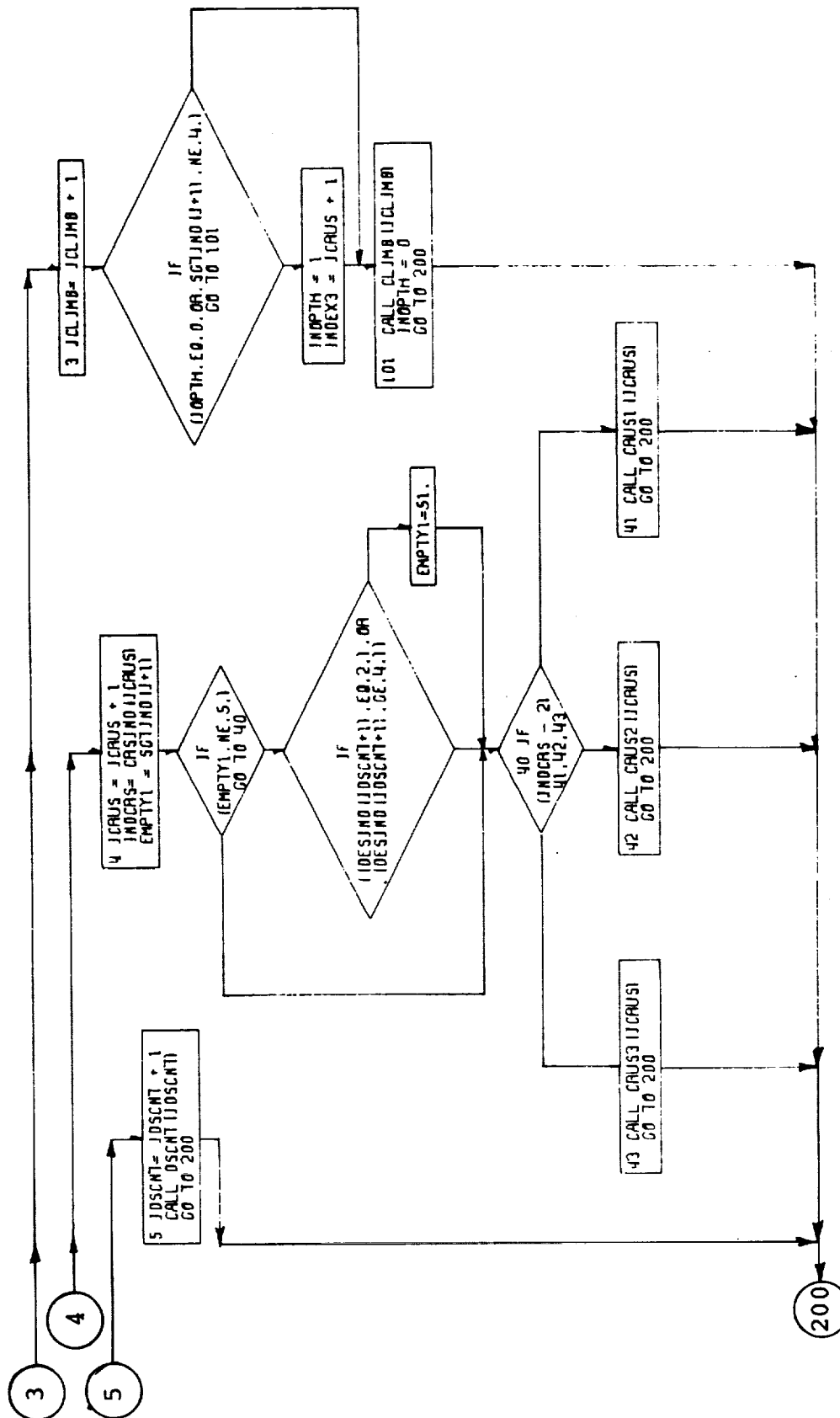


Figure 4-41. Performance Calculations Subprogram Flow Chart (Part 4 of 5).



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4.12.1 Taxi Calculations Subroutine

The taxi calculations subroutine (specified by SGTIND = 1), calculates the fuel required to taxi at ground idle engine setting for a specified period of time. For aircraft which have independent auxiliary cruise propulsion systems (AIPIND = 2), the program will calculate taxi performance for either primary engines operating alone, or both primary and auxiliary cruise propulsion engines operating. This is accomplished by means of the input constant k_{FI} . If $k_{FI} = 0$, the program will consider only primary engines in operation in determining fuel flow rates. If $k_{FI} = 1$, the program will include both primary and lift propulsion systems in calculating the fuel flow rates and the corresponding reduction in aircraft gross weight. Figure 4-42 is a flow chart of this subroutine.

Input to this subroutine consists of the time for taxi, value of k_{FI} , and atmospheric conditions.

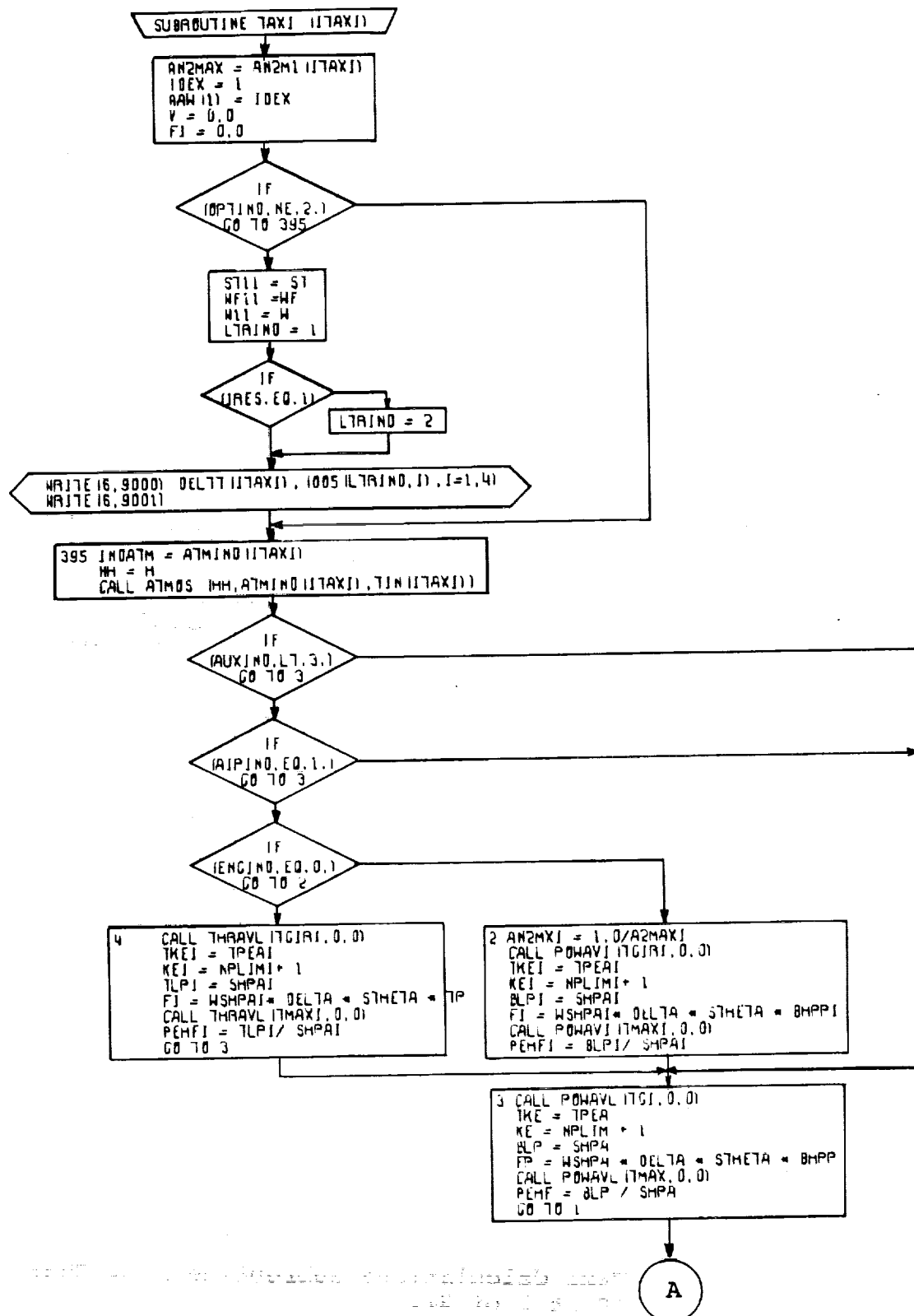


Figure 4-42. Taxi Calculations Subroutine Flow Chart
(Part 1 of 2).

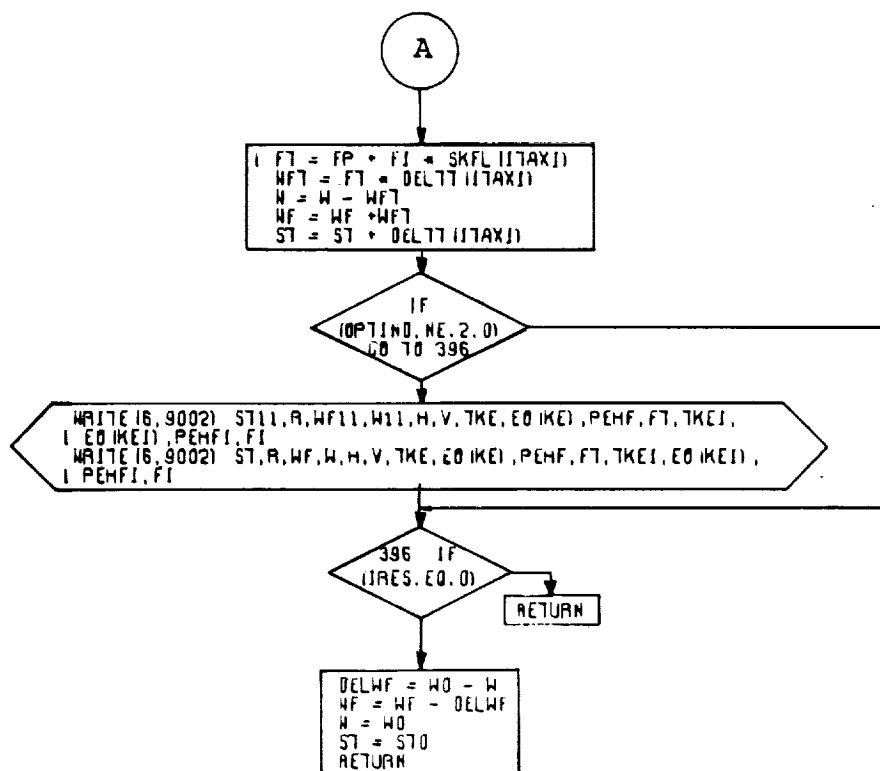


Figure 4-42. Taxi Calculations Subroutine Flow Chart
(Part 2 of 2).

4.12.2 Takeoff, Hover, and Landing Calculations Subroutine

The takeoff, hover, and landing calculations subroutine (specified by SGTIND = 2) will calculate the thrust or power required and corresponding fuel flow rates during simulated takeoff/hover/landing operations. Four options are available, specified by the input indicator TOLIND:

- TOLIND = 1 - Input required thrust-weight ratio and vertical rate of climb. Program will calculate required power fractions.
- TOLIND = 2 - Input the required power fraction and vertical rate of climb. Program will calculate thrust-weight ratio.
- TOLIND = 3 - This option is the same as TOLIND = 1, except hover in ground effect is assumed, requiring the input of height of fuselage bottom above ground as a fraction of main rotor diameter.
- TOLIND = 4 - This option is the same as TOLIND = 2, except hover in ground effect is assumed, requiring the input of height of fuselage bottom above ground as a fraction of main rotor diameter.

In all cases, the program will print out the power fraction and thrust-weight ratio. The program will permit operation at power fractions greater than 1.0 (more than 100 percent of available power) in order to make it easier to perform studies in which engine power is being varied parametrically to satisfy specified takeoff or landing requirements as a site. The program will, however, print a cautionary note that power fraction exceeds 100 percent. In the case (TOHL = 2 or 4) where the required power fraction is input, if the calculated thrust-weight ratio is less than the design thrust-weight ratio input (LOC 0228), the following cautionary note will be printed: INSUFFICIENT POWER AVAILABLE TO HOVER. T/W AVAILABLE LESS THAN T/W REQUIRED AT DESIGN DOWNLOAD.

For a helicopter configuration having auxiliary independent engines, the program sets the auxiliary engine power setting at ground idle.

It is possible to use a hover segment in the mission profile to account for a reserve fuel requirement (SGTIND = 20), in such a case the helicopter weight at the end of hover is set back to the weight at the beginning of hover, or as a part of the basic mission (in this case the weight is not reset). In either case, the fuel used during hover is included in the total fuel required to size the helicopter.

Figure 4-43 is a flow chart for this subroutine.

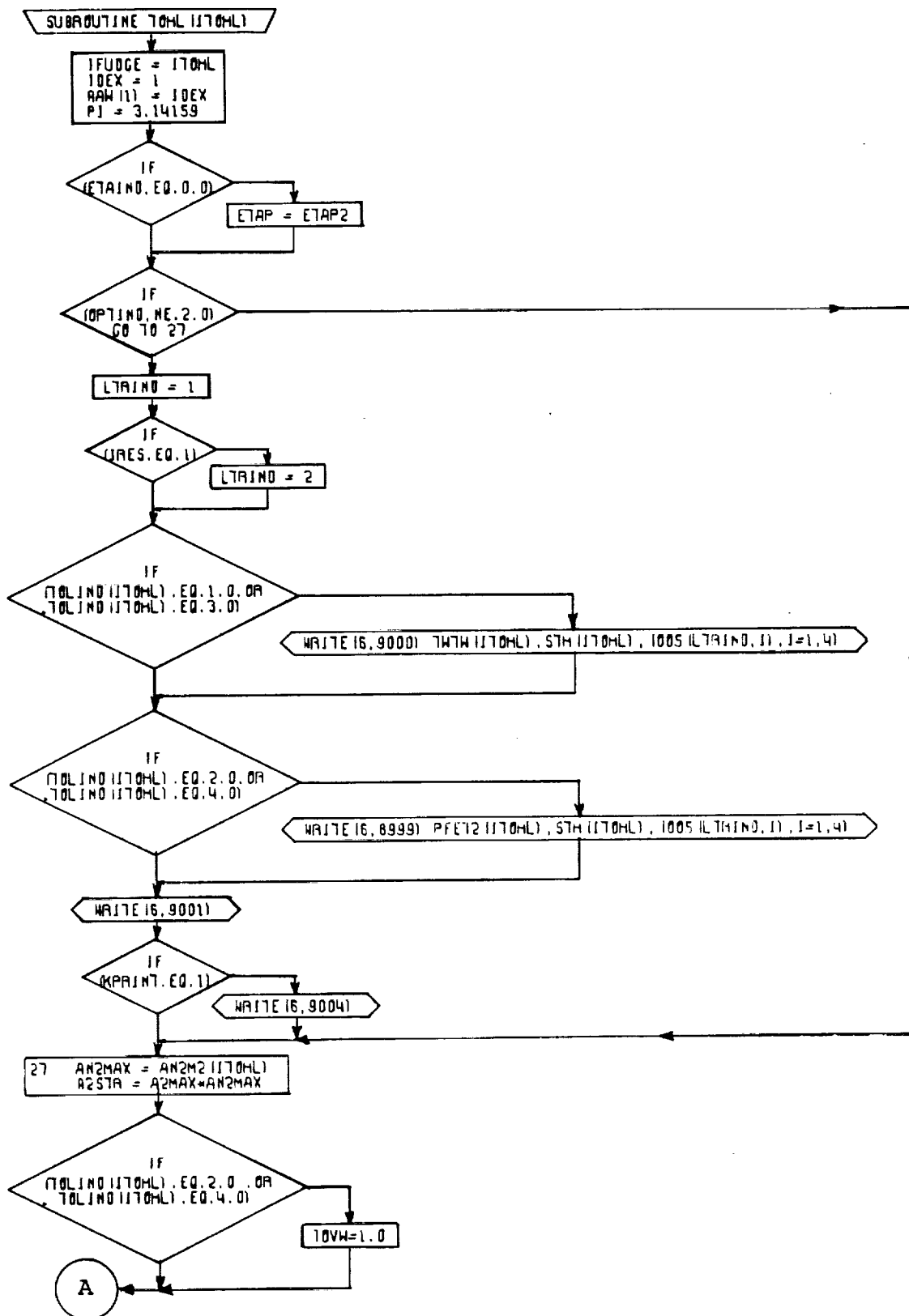


Figure 4-43. Takeoff, Hover, and Landing Subroutine Flow Chart (Part 1 of 4).

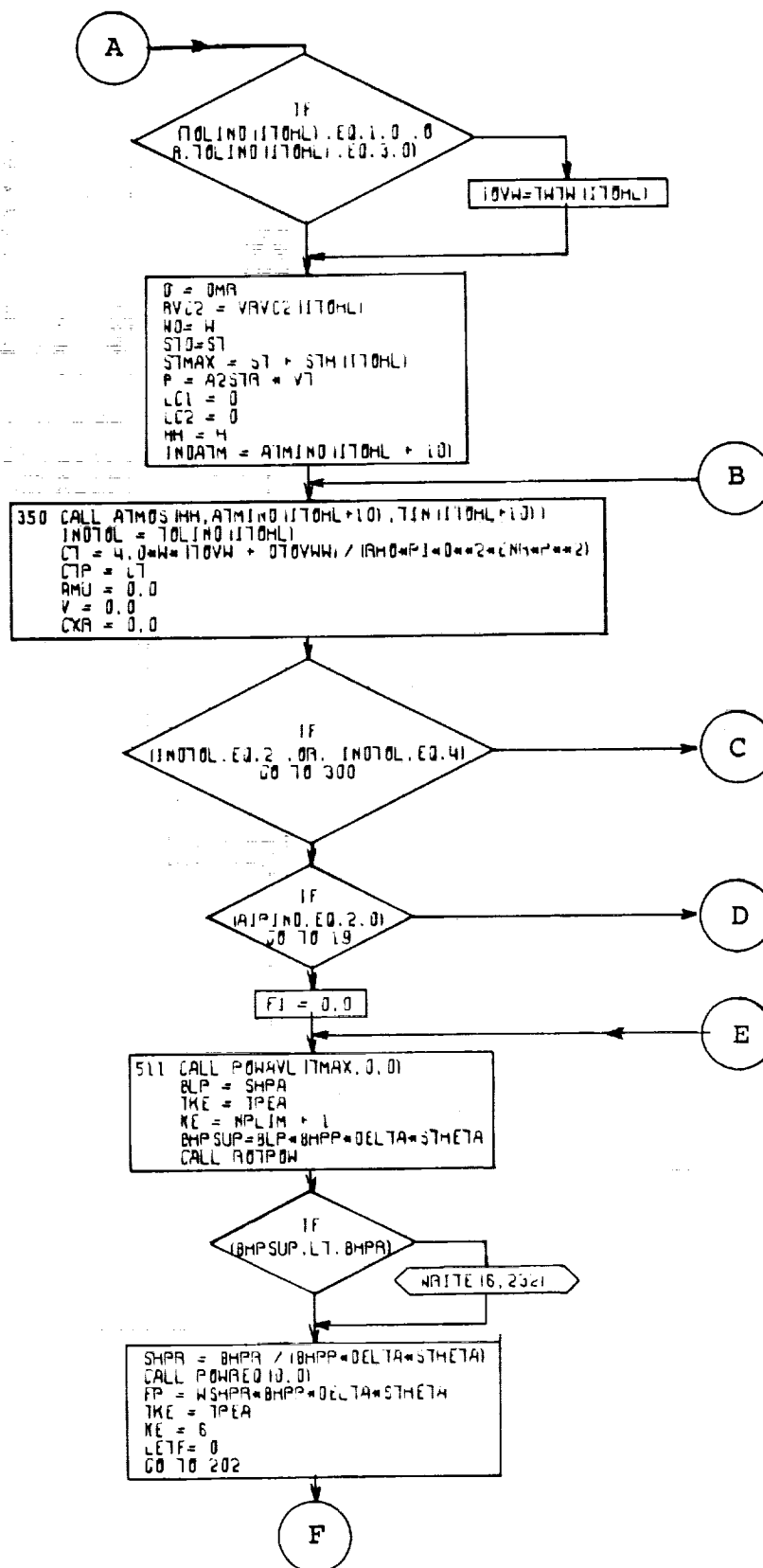


Figure 4-43. Takeoff, Hover, and Landing Subroutine Flow Chart (Part 2 of 4).

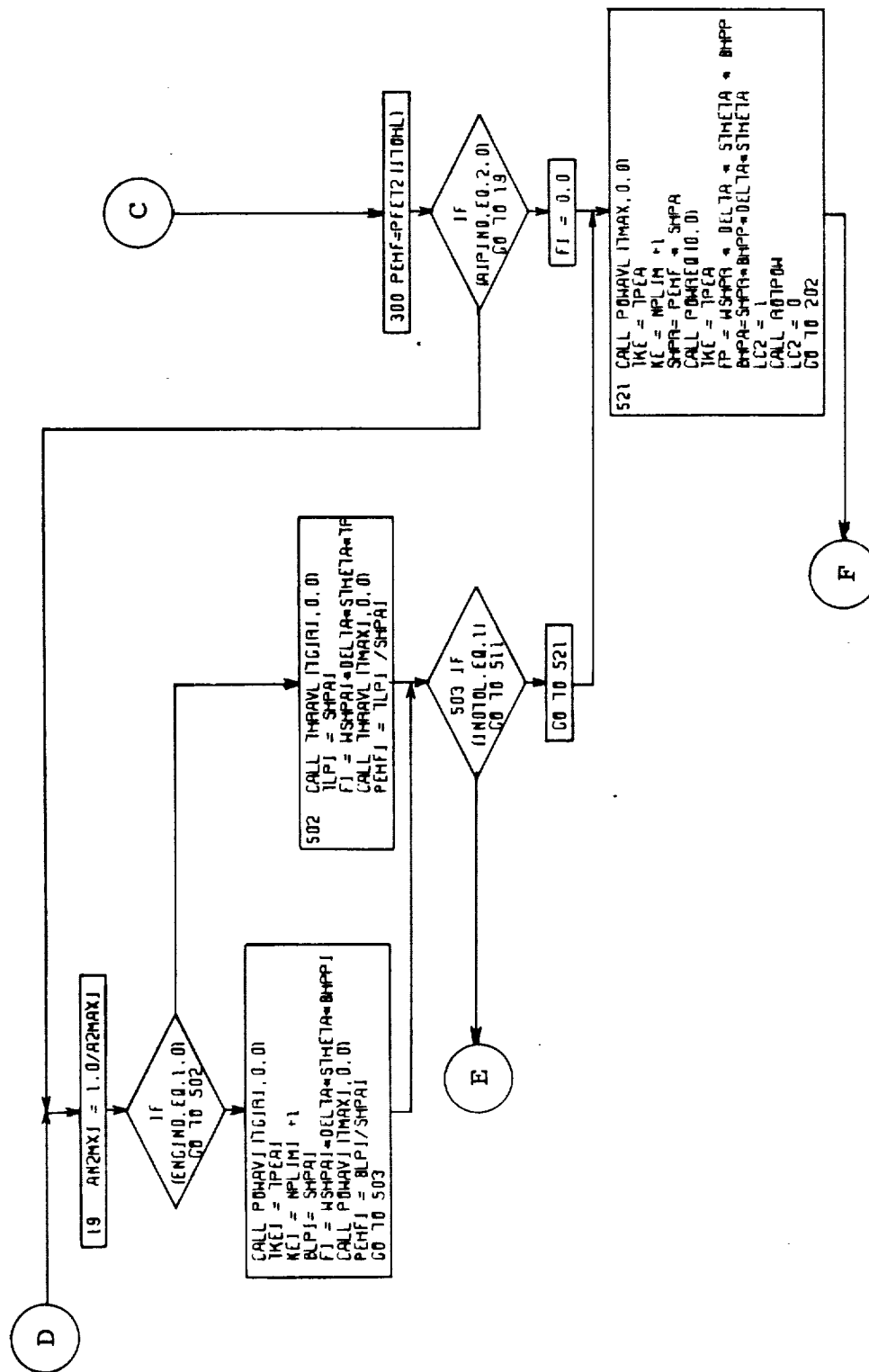


Figure 4-43. Takeoff, Hover, and Landing Subroutine
Flow Chart (Part 3 of 4).

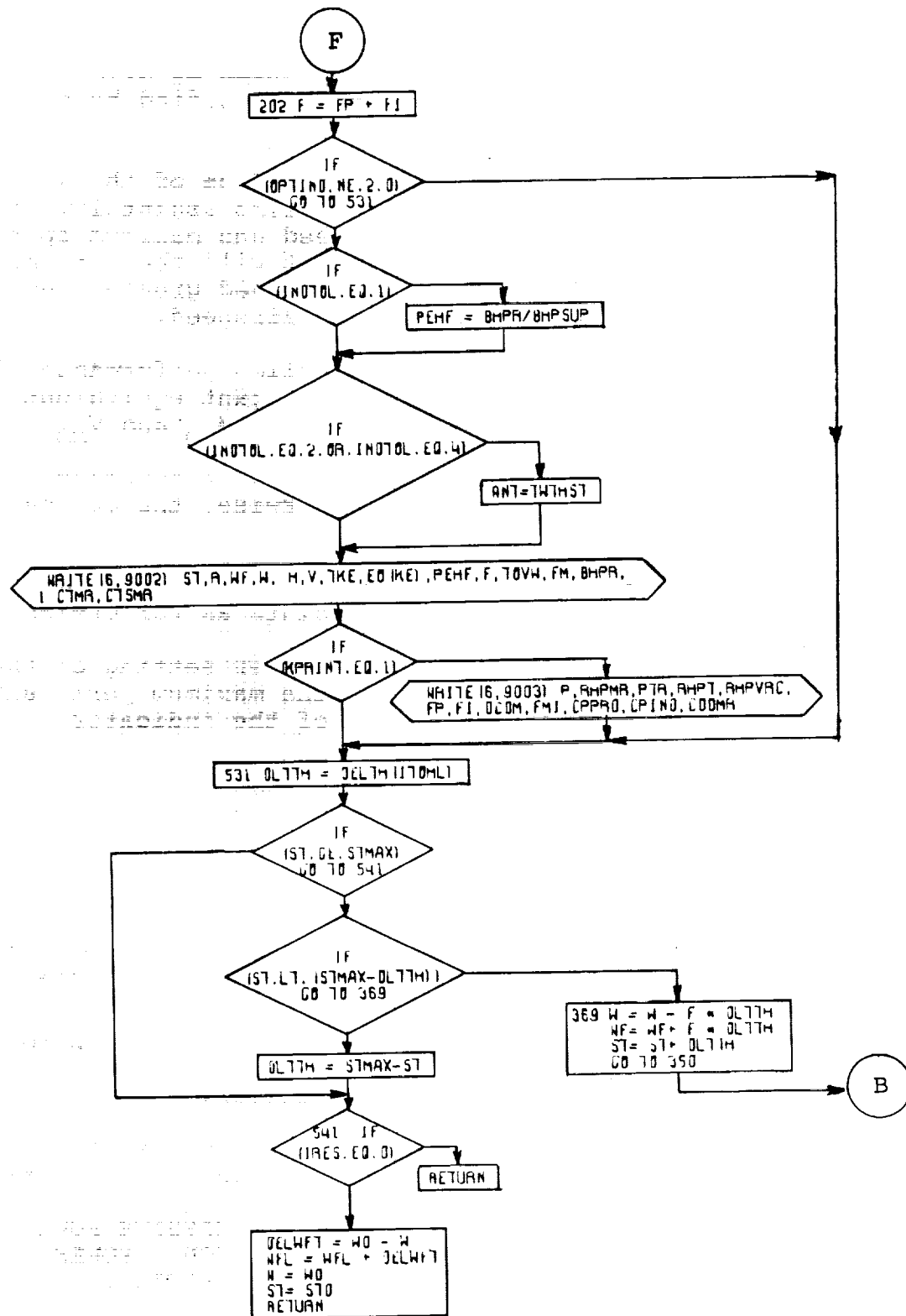


Figure 4-43. Takeoff, Hover, and Landing Subroutine
Flow Chart (Part 4 of 4).

4.12.3 Climb Calculations Subroutine

The third performance segment is a calculation of climb performance. Four options are available, specified by the indicator CLMIND:

- CLMIND = 1 - The program calculates performance of the aircraft in a maximum rate of climb ascent limited by maximum operating airspeed and maximum operating Mach number. In no event will the aircraft be required to fly at an airspeed greater than the input maximum operating airspeed.
- CLMIND = 2 - The program calculates the climb performance of the aircraft at specified constant equivalent airspeed limited, as before, by M_{MO} and V_{MO} .
- CLMIND = 3 - Climb performance is calculated at constant specified Mach number. Otherwise, the option is similar to CLMIND = 2.
- CLMIND = 4 - Climb will be calculated at constant true airspeed with the same constraints as for CLMIND = 2.

For all options, the user may input the power setting of the engines which will be considered to be the maximum permissible rating. This is accomplished by means of the indicator POWIND:

POWIND = 0: Maximum	} engine rating
POWIND = 1: Military	
POWIND = 2: Normal	

The user may specify a value for incremental equivalent flat plate area parasite drag during climb, ΔF_{eCLIMB} , to represent variations in store drag.

Engine shutdown during climb may be simulated by inputs for N_{psd} (primary engines) and N_{psdi} (auxiliary independent engines). One or more engines may be shutdown.

If the flight path (climb) angle exceeds 90 degrees, the engine power setting is reduced and the program prints out:

CAUTION: CLIMB ANGLE TOO LARGE DUE TO EXCESSIVE POWER
AVAILABLE AT THIS FLIGHT CONDITION. POWER
SETTING REDUCED TO _____ ENGINE RATING.

If there is insufficient power available for climb, the engine power setting is increased and the program prints out:

CAUTION: INSUFFICIENT POWER AVAILABLE FOR CLIMB AT
THIS FLIGHT CONDITION. POWER SETTING
INCREASED TO _____ ENGINE RATING.

The input h_{max} has two applications. If $h_{OPTIND} = 1$ (optimum altitude search) and the climb is followed by a cruise, the input value of h_{max} will be interpreted as the maximum flight altitude for the following cruise. If the optimum cruise altitude is determined by the program to be at an altitude less than h_{max} , the climb will terminate at the lower altitude. If an optimum altitude search is not being used or if the following segment is other than a cruise, the input h_{max} is interpreted as the final altitude for the climb segment.

It is possible to use a climb segment in the mission profile to account for a reserve fuel requirement ($SGTIND = 30$) (in such a case the helicopter weight at the end of climb is set back to the weight at the beginning of climb) or as a part of the basic mission (in this case the weight is not reset). In either case, the fuel used during climb is included in the total fuel required to size the helicopter.

Figure 4-44 is a flow chart for this subroutine.

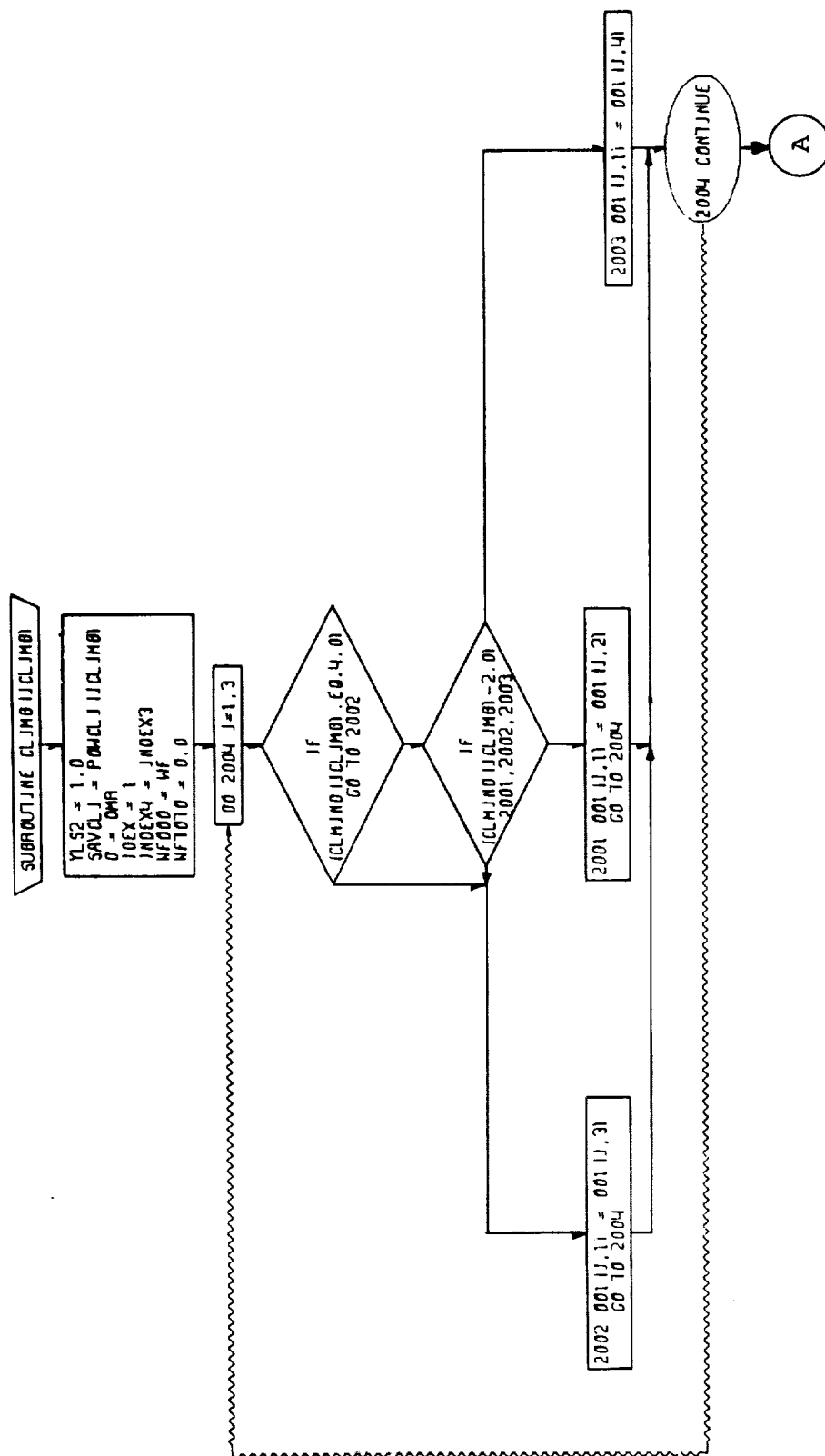


Figure 4-44. Climb Calculations Subroutine Flow Chart (Part 1 of 13).

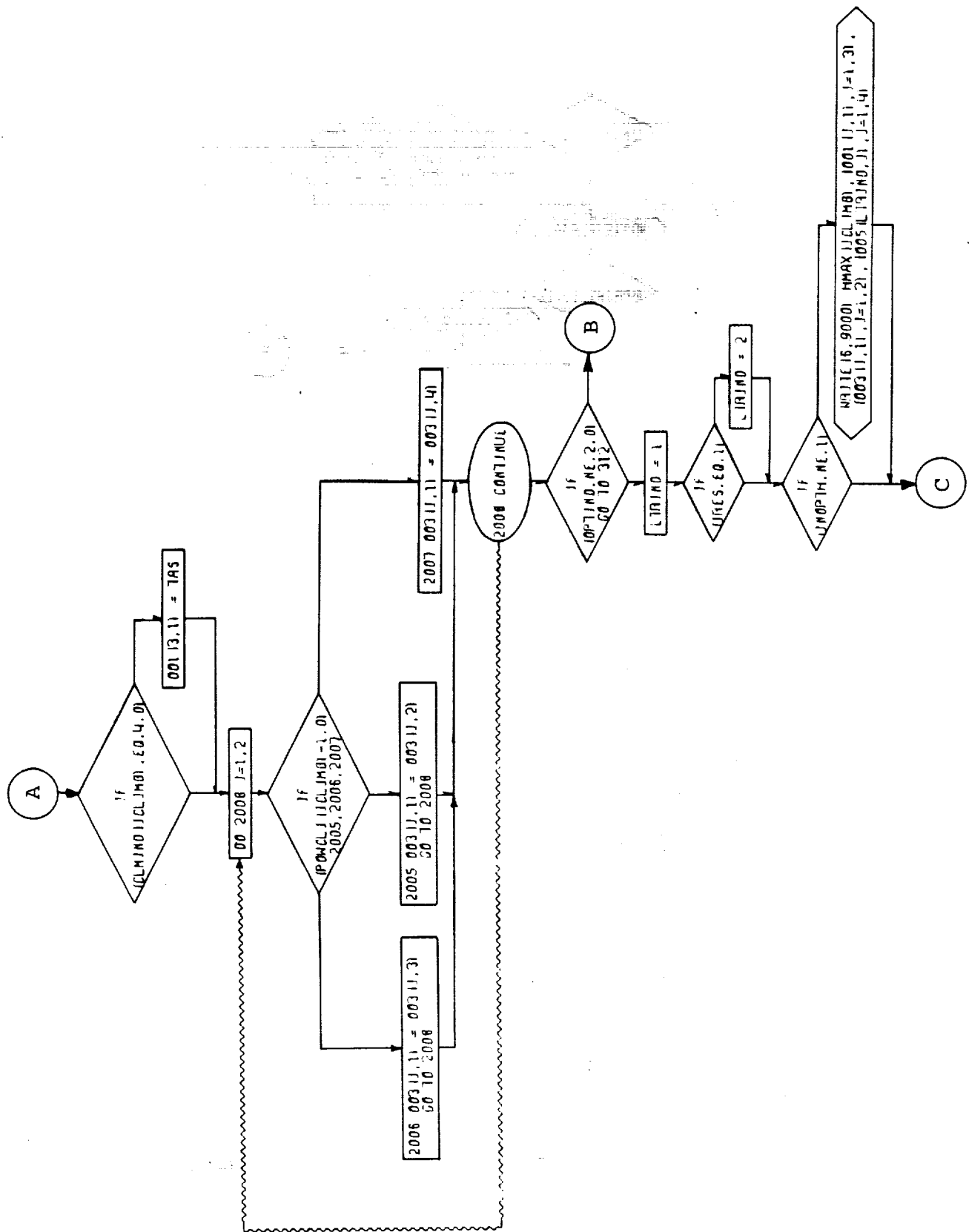


Figure 4-44. Climb Calculations Subroutine Flow Chart (Part 2 of 13).

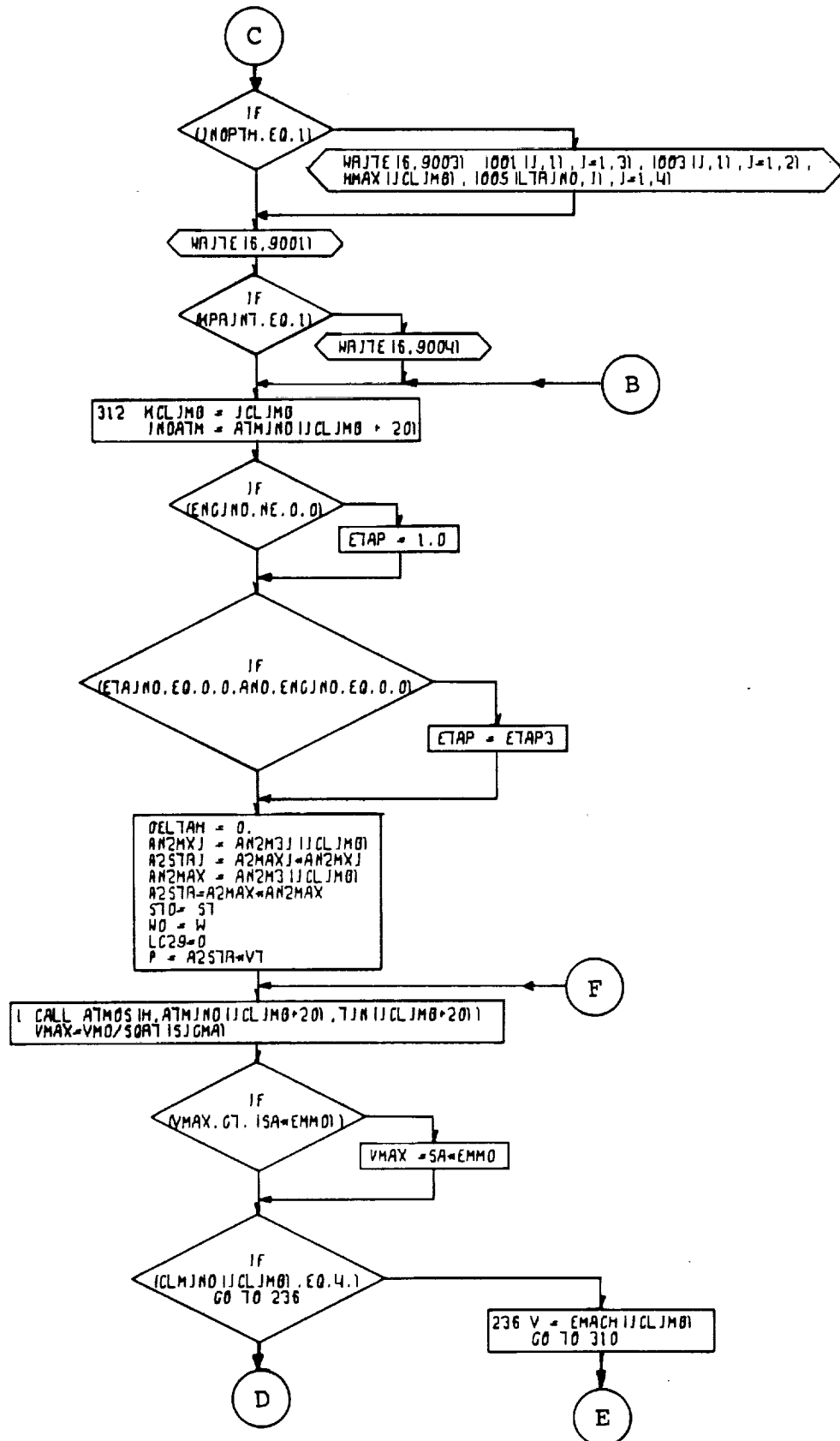


Figure 4-44. Climb Calculations Subroutine Flow Chart
(Part 3 of 13).

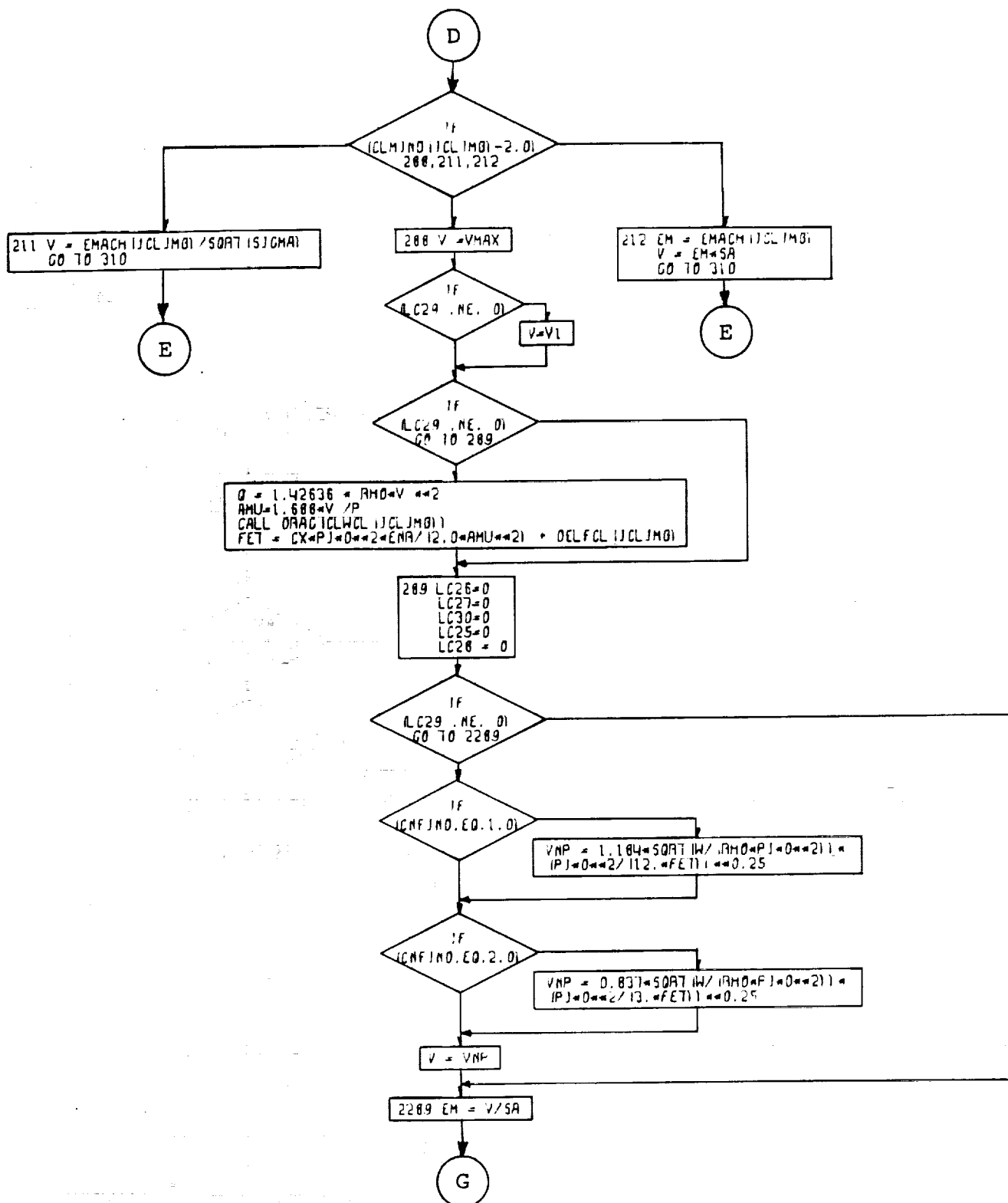


Figure 4-44. Climb Calculations Subroutine Flow Chart
(Part 4 of 13).

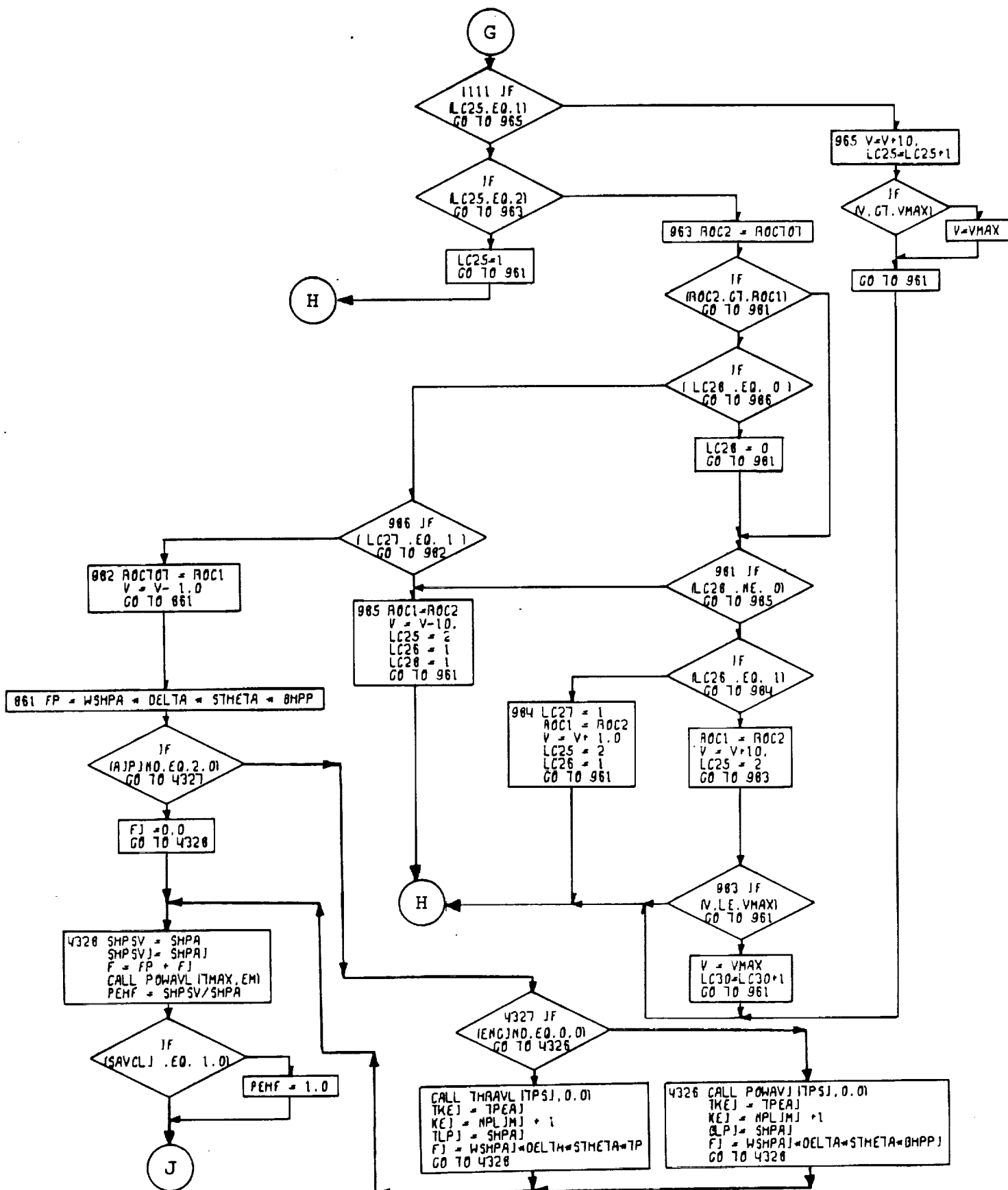


Figure 4-44. Climb Calculations Subroutine Flow Chart
(Part 5 of 13).

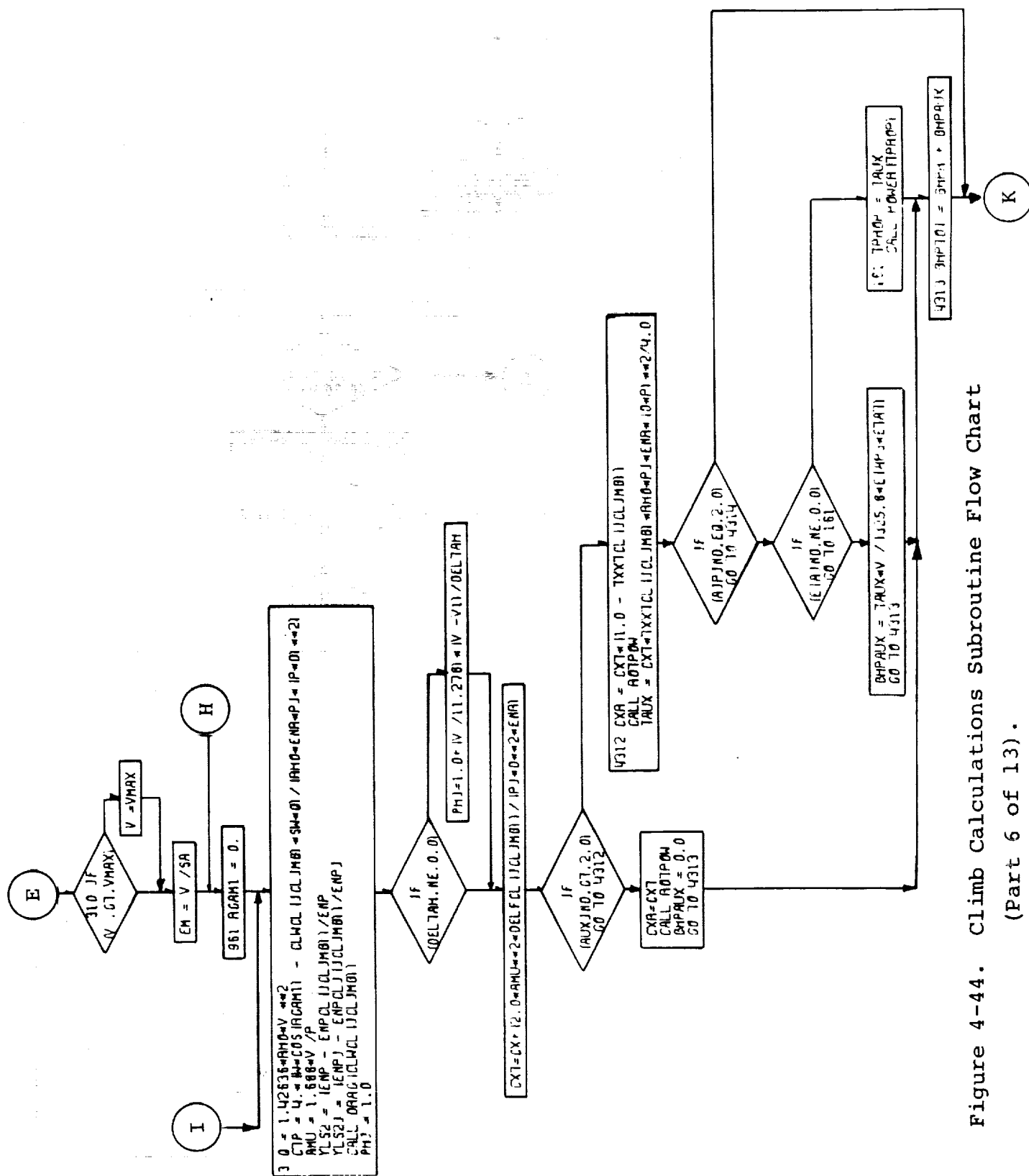


Figure 4-44. Climb Calculations Subroutine Flow Chart
(Part 6 of 13).

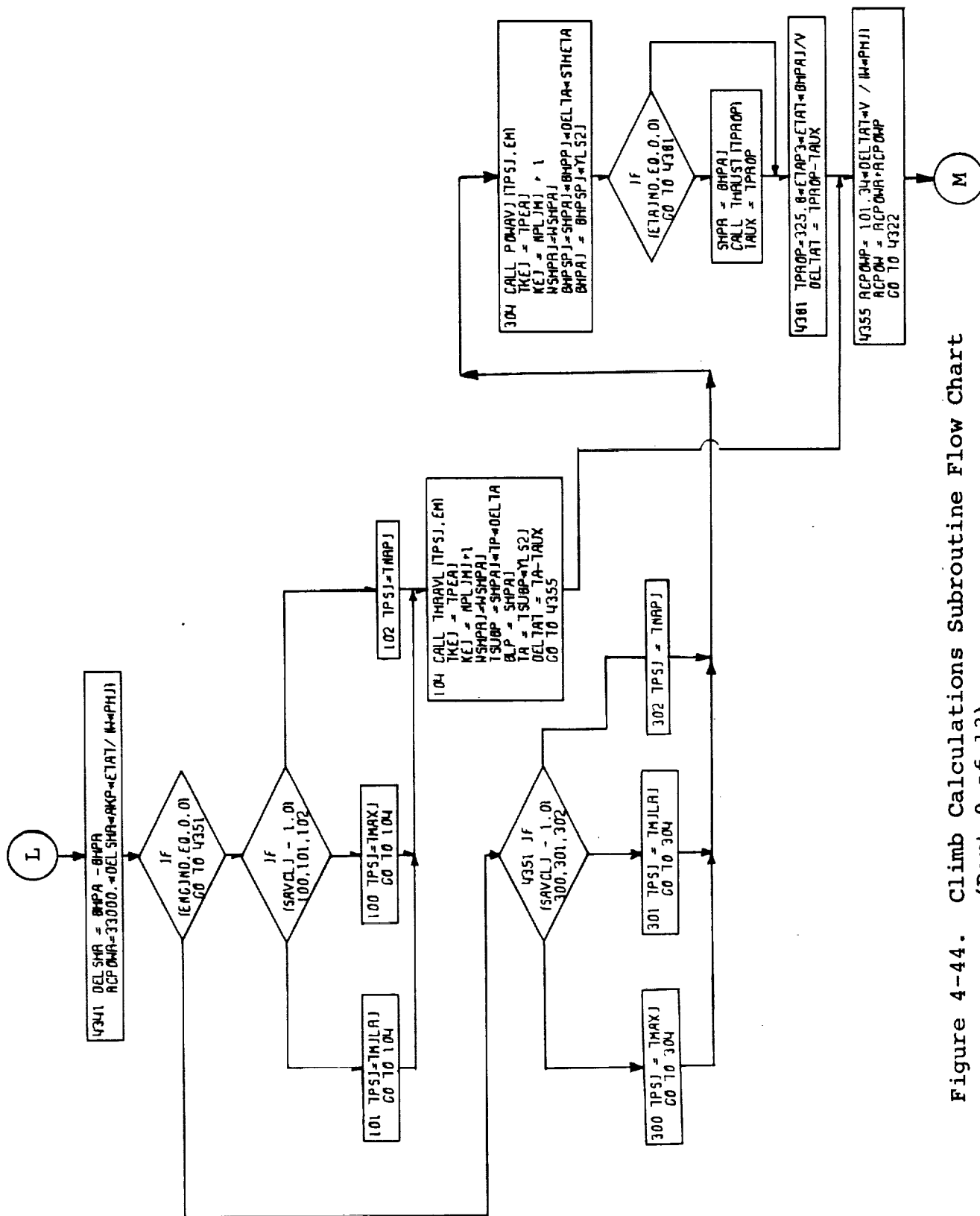


Figure 4-44. Climb Calculations Subroutine Flow Chart
(Part 9 of 13).

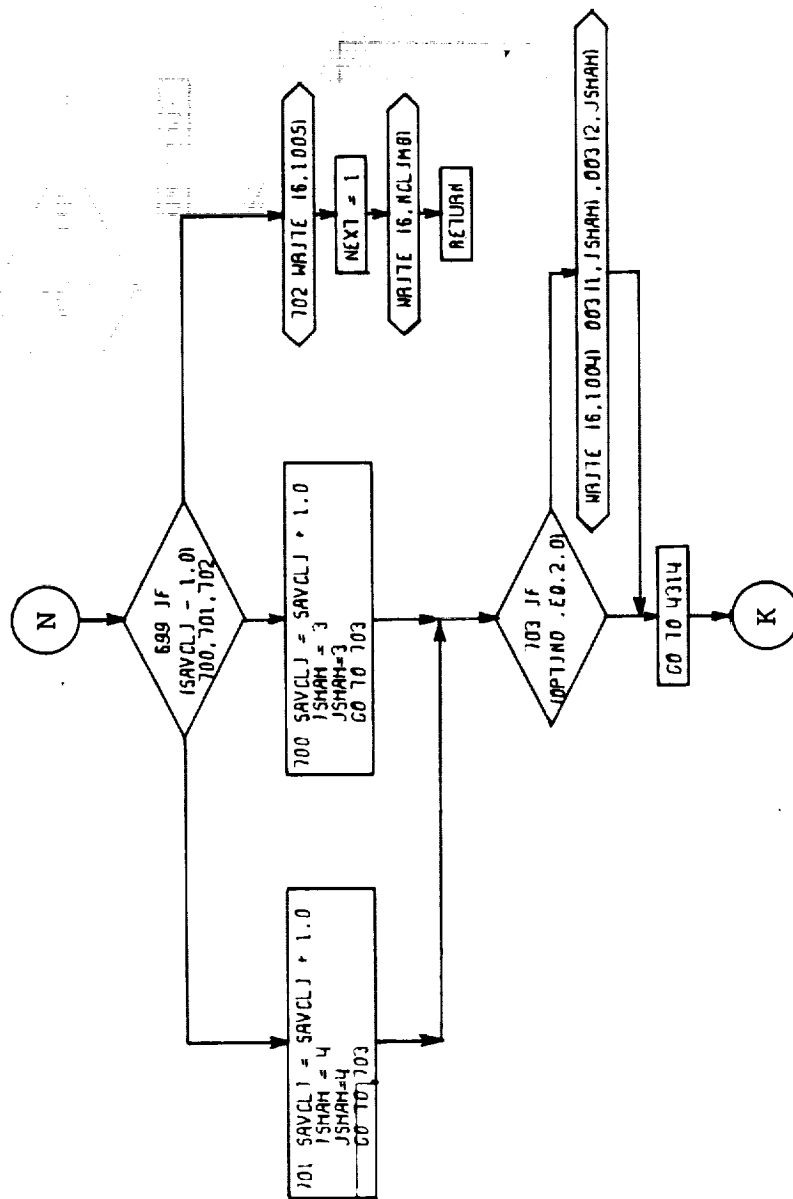


Figure 4-44. Climb Calculations Subroutine Flow Chart (Part 10 of 13).

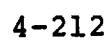


Figure 4-44. Climb Calculations Subroutine Flow Chart (Part 11 of 13).

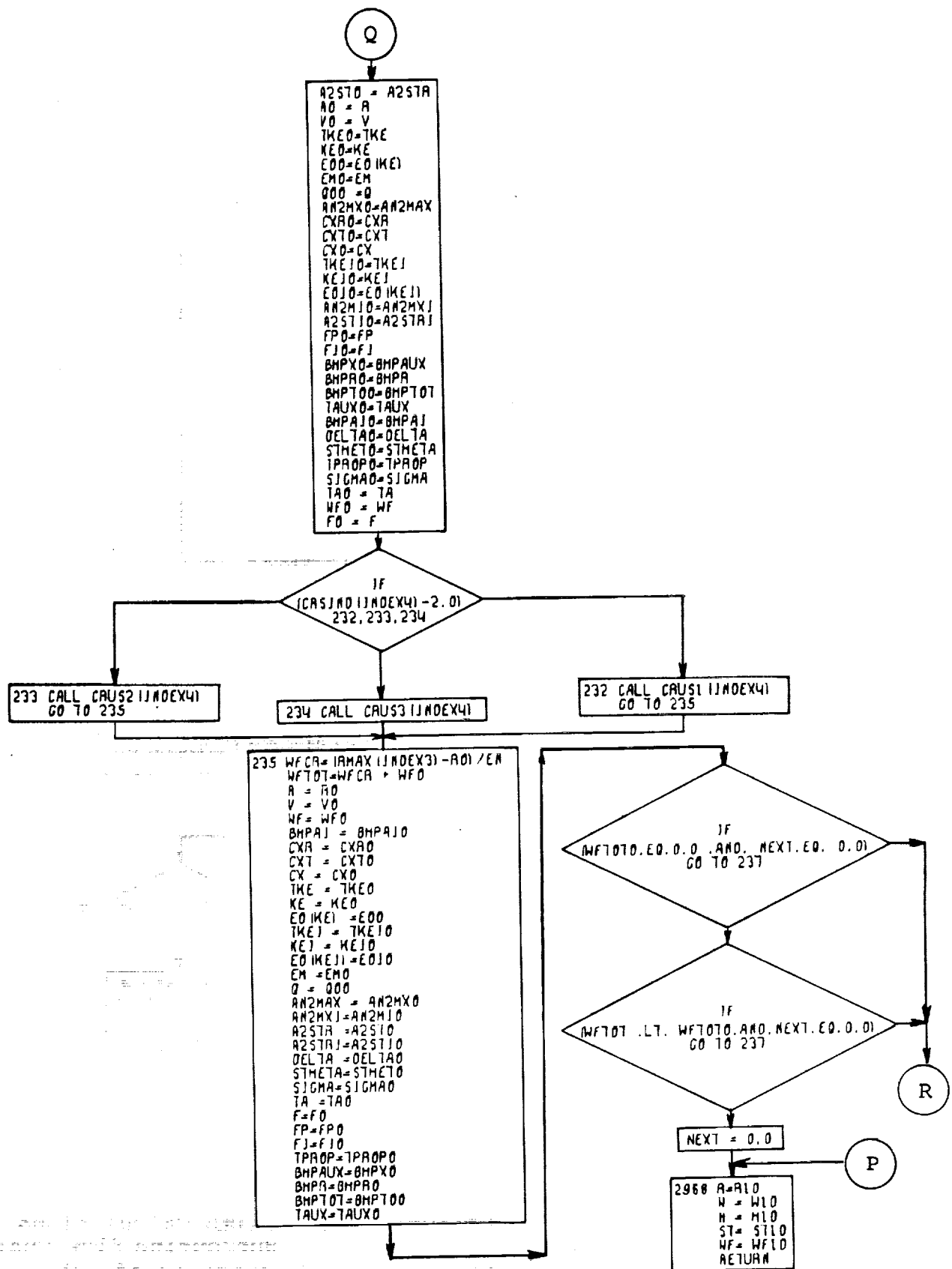


Figure 4-44. Climb Calculations Subroutine Flow Chart
(Part 12 of 13).

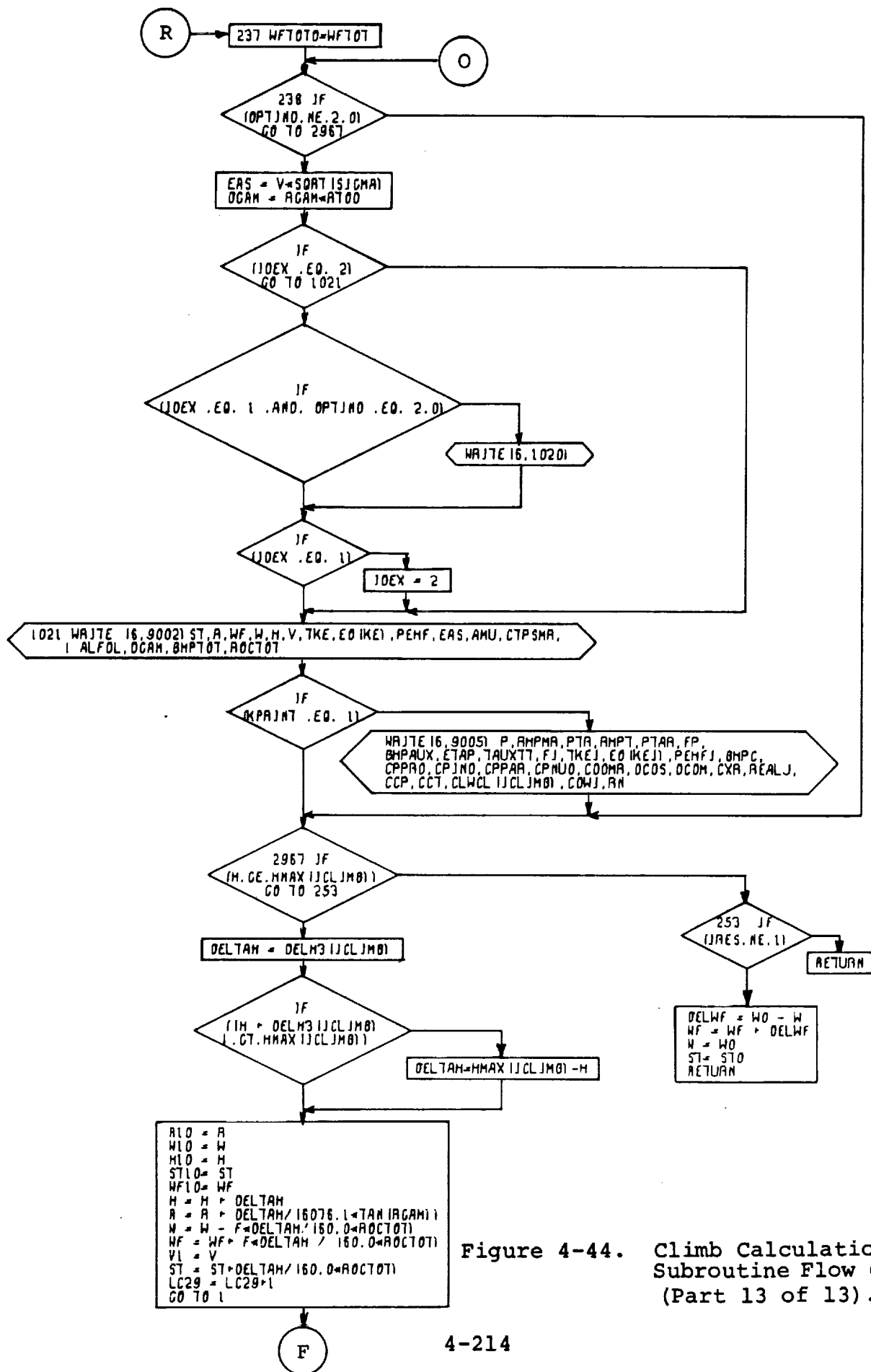


Figure 4-44. Climb Calculations Subroutine Flow Chart (Part 13 of 13).

4.12.4 Cruise Calculations Subroutine

The fourth performance segment is the calculation of cruise performance. The cruise performance calculation contains six separate options specifying the type of cruise for the aircraft. This option is determined by an input idnicator, CRSIND.

- CRSIND = 1 - This is a calculation of helicopter cruise performance at a fixed cruise power setting and at a constant altitude, constrained by limiting airspeed and Mach number. This option calculates the true airspeed, helicopter advance ratio, specific range, and reduction in gross weight during cruise. In the case of compound and auxiliary propulsion helicopters, if the auxiliary propulsion power required (to satisfy the input T_{AUX}/T_{TOT}) is greater than that available, as determined by POWIND, T_{AUX}/T_{TOT} is readjusted to match the power requirements. It should be further noted that in the case of a configuration having auxiliary independent engines POWIND, which specifies the desired power setting for the primary engines, is used as a limiting factor for the auxiliaries.
- CRSIND = 2 - This option will calculate the cruise performance constrained by cruise power and by limiting airspeed and Mach number of the aircraft at constant true airspeed and constant altitude. The program will calculate the power setting required, true airspeed, specific range, and corresponding reduction in gross weight of the aircraft during cruise. In the case of an auxiliary independent engine configuration, if either the primary or auxiliary engine power required is greater than that available, T_{AUX}/T_{TOT} is readjusted accordingly. Then, if a power required-power available match is not achieved, cruise speed is reduced.
- CRSIND = 3 - This option calculates the airspeed during cruise required for best specific range, constrained by normal power setting and by limiting airspeed and Mach number. Flight is at constant altitude. When auxiliary independent engines are employed, cruise speed is reduced (T_{AUX}/T_{TOT} not being readjusted from its input value) until both auxiliary and primary power required are less than available.
- CRSIND = 4 - This option will calculate the "long range cruise" condition; that is, cruise at speed for

99% of best specific range. Flight is constrained by normal power setting, limiting airspeed and Mach number and is at constant altitude.

CRSIND = 5 - This option is a calculation for a cruise-climb at a constant value of W/δ (airplane weight to ambient pressure ratio). The airspeed will be the speed for best specific range.

CRSIND = 6 - This is a calculation for a cruise climb (constant W/δ) at the speed for 99% of best specific range.

Cruise power setting as discussed above is defined by user input to be maximum (POWIND = 0), military (POWIND = 1), or normal (POWIND = 2) engine rating. This subroutine permits simulation of cruise performance of an aircraft with an arbitrary number of engines (both primary and auxiliary) shut down.

The program user specifies the number of engines shut down and a corresponding increment in airplane equivalent flat plate area drag.

The user may also specify a desired value of headwind when CRSIND = 3 through 6.

It is possible to use a cruise segment in the mission profile to account for a reserve fuel requirement (SGTIND = 40) (in such a case the helicopter weight at the end of cruise is set back to the weight at the beginning of cruise) or as a part of the basic mission (in this case the weight is not reset). In either case, the fuel used during cruise is included in the total fuel required to size the helicopter.

The input for the subroutine consists of the final range for cruise, the step size (incremental range), number of engines shut down, increment in drag coefficient, atmospheric conditions, required true airspeed (if CRSIND = 2) the headwind (if CRSIND = 3, 4, 5, or 6) and the settings for CRSIND and POWIND. Figure 4-45 is a flow chart of this subroutine.

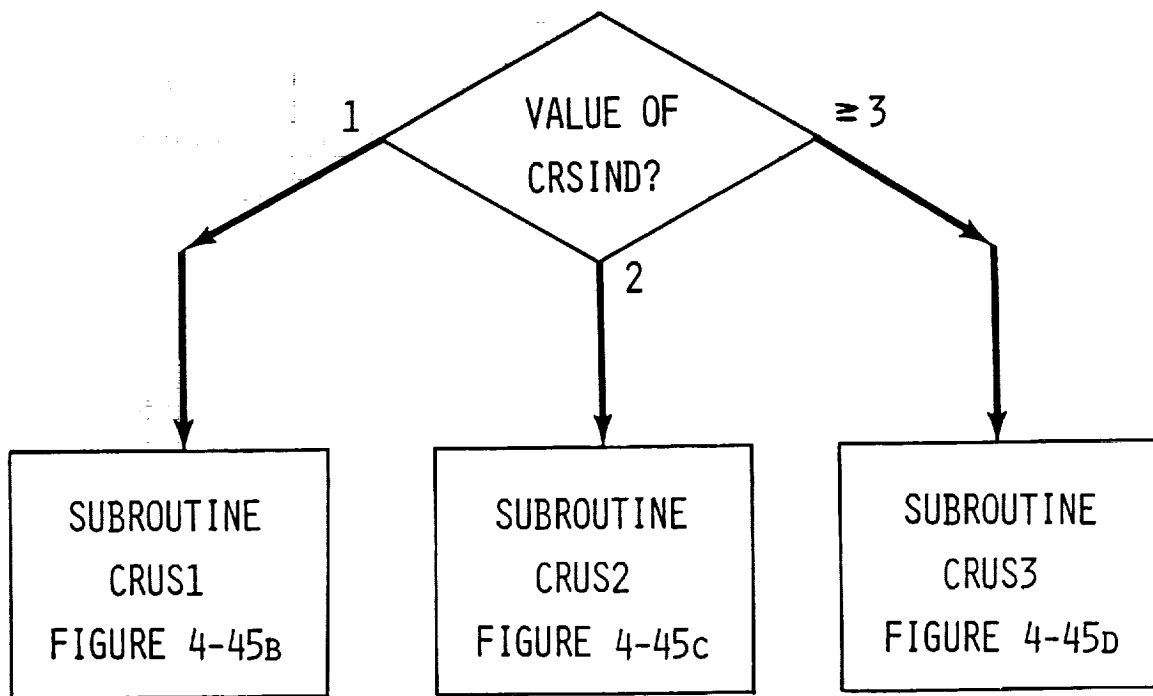


Figure 4-45a. Cruise Calculations Subroutine.

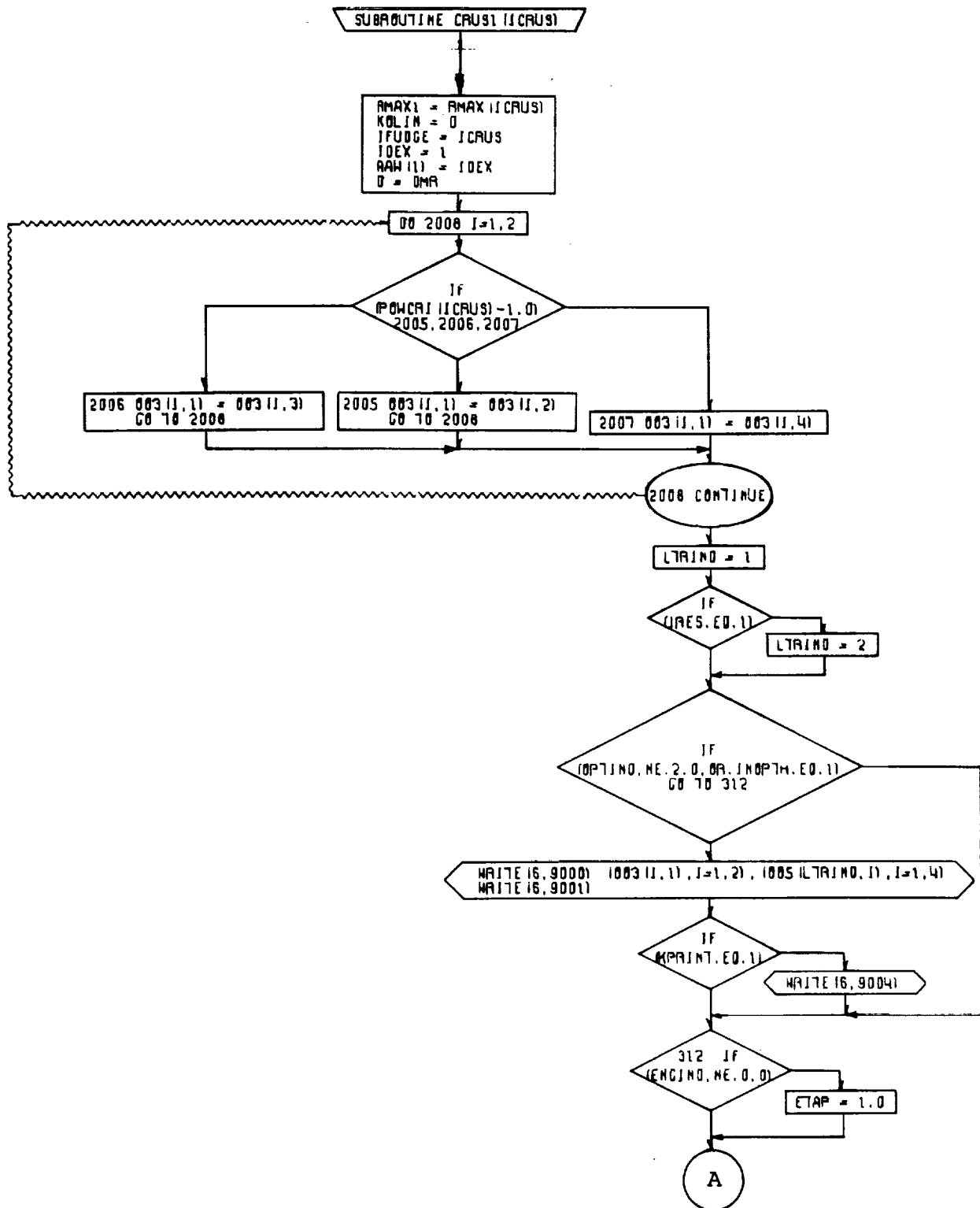


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 1 of 8).

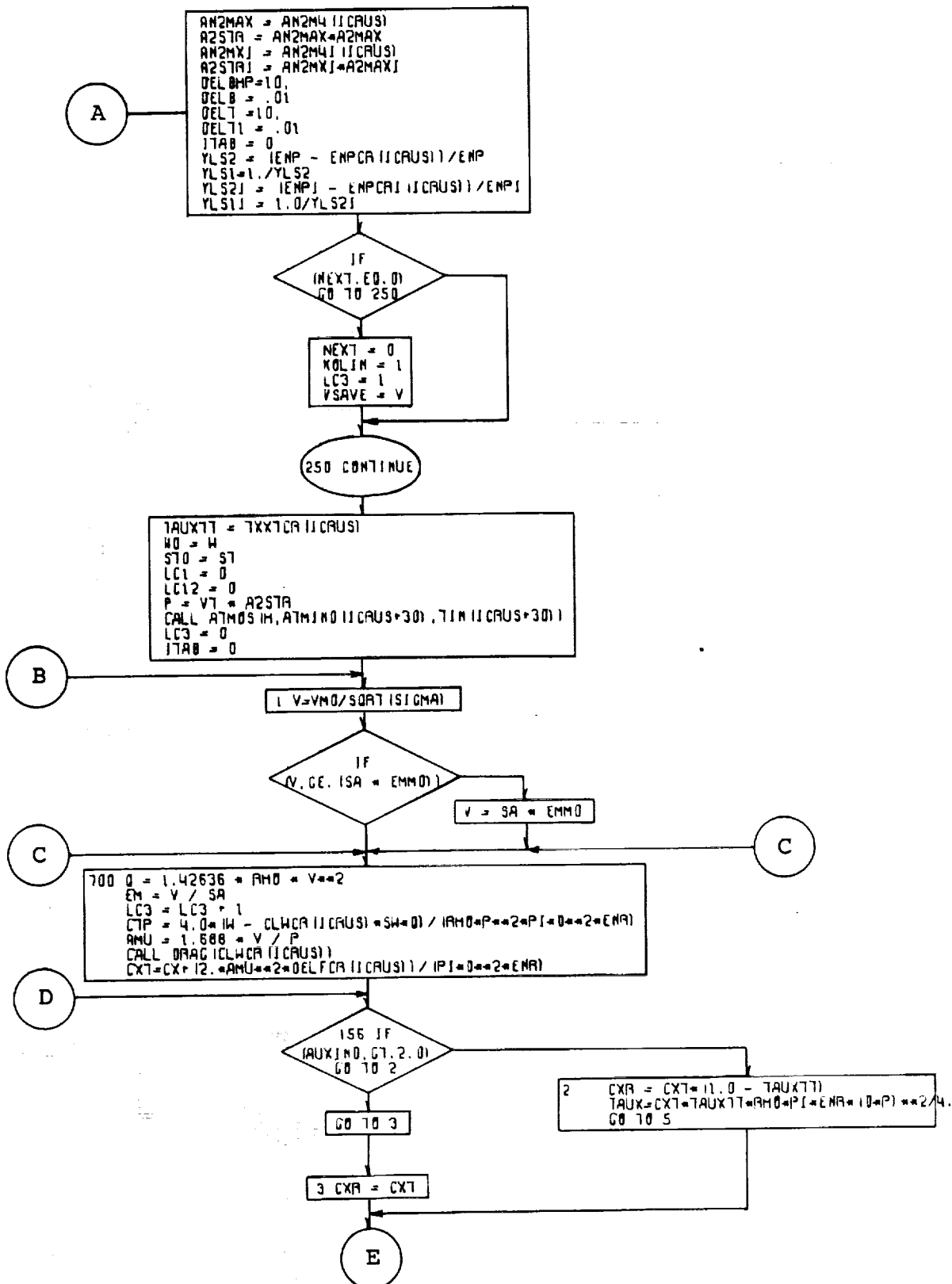


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 2 of 8).

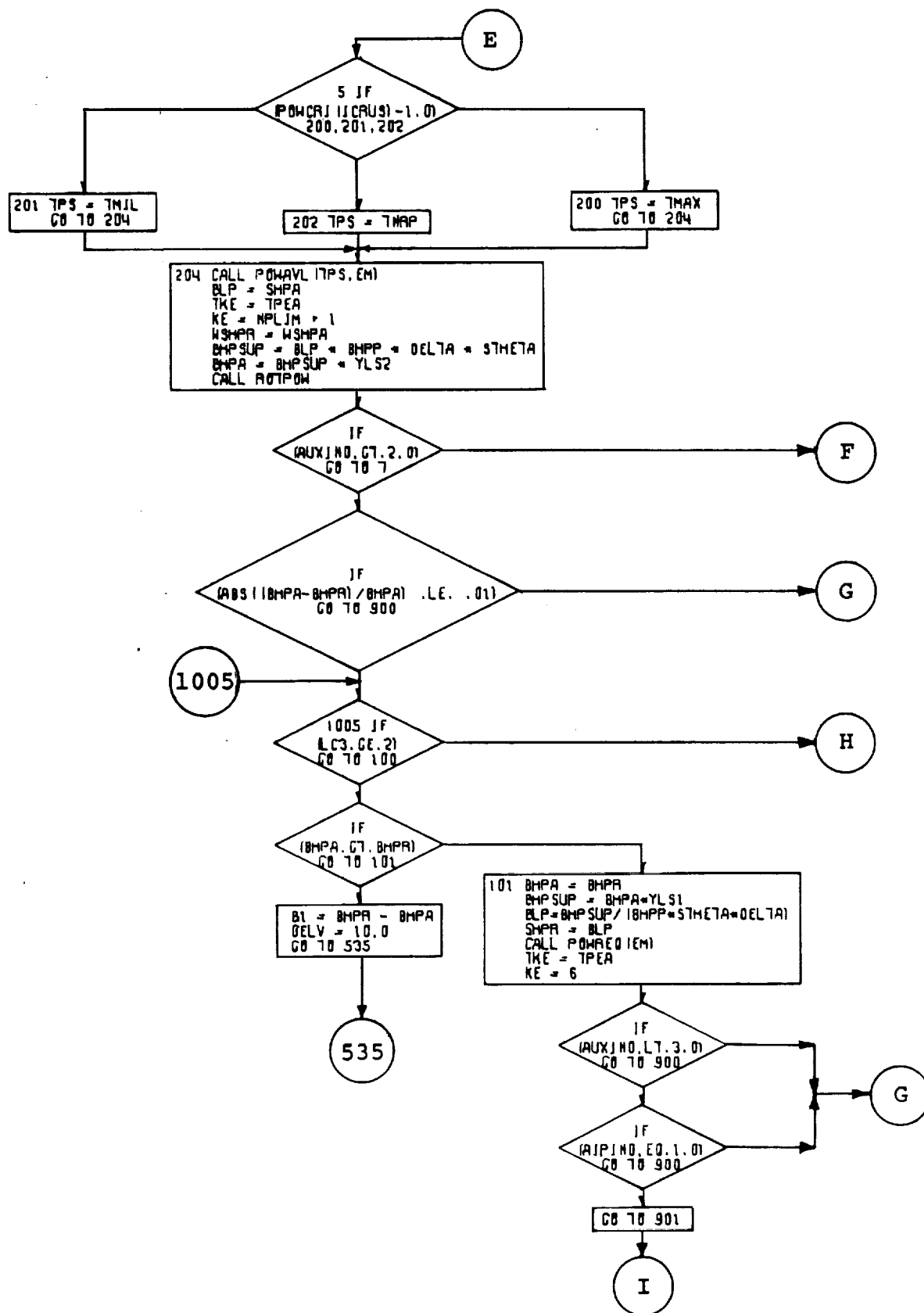


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 3 of 8).

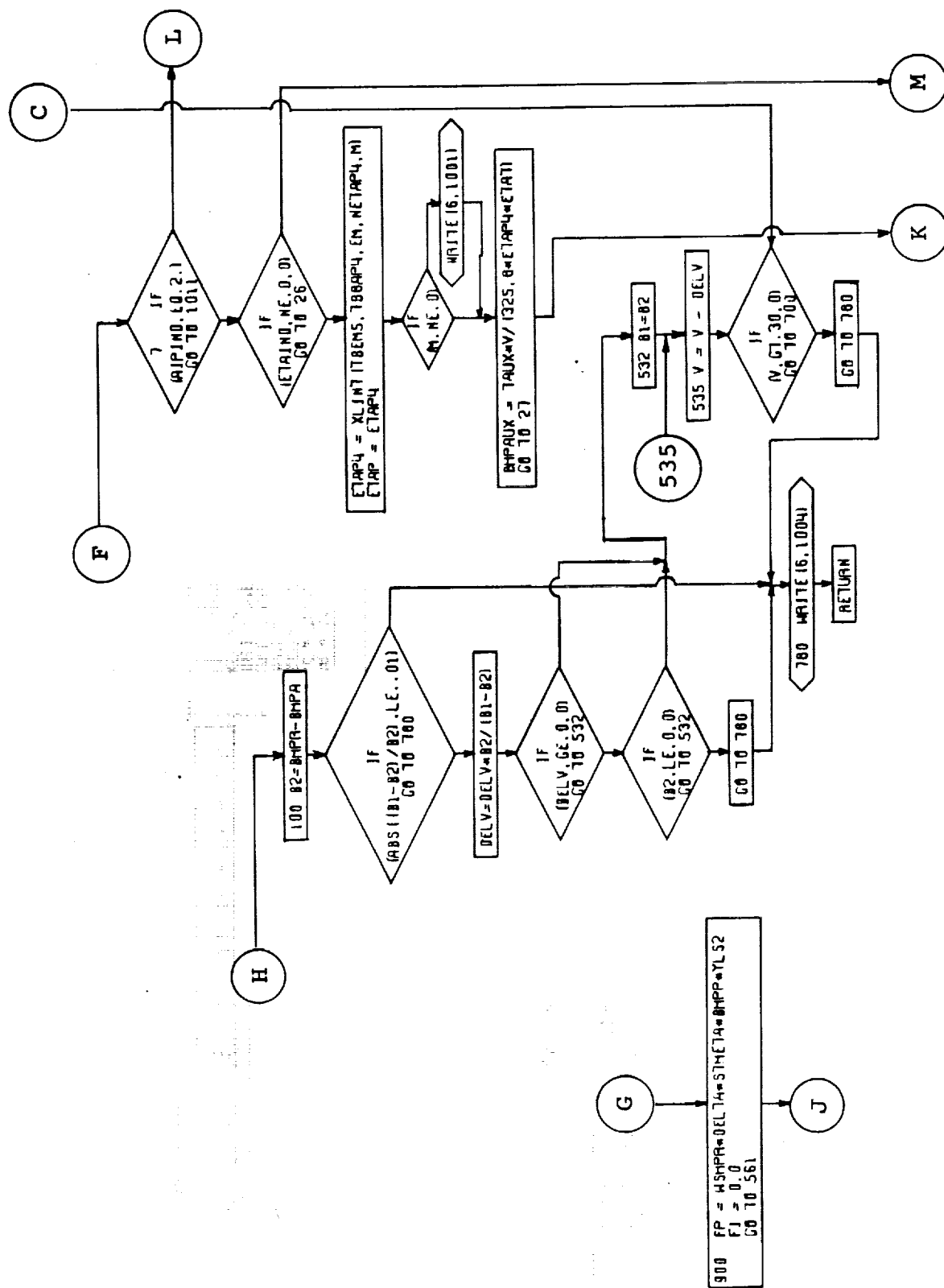


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 4 of 8).

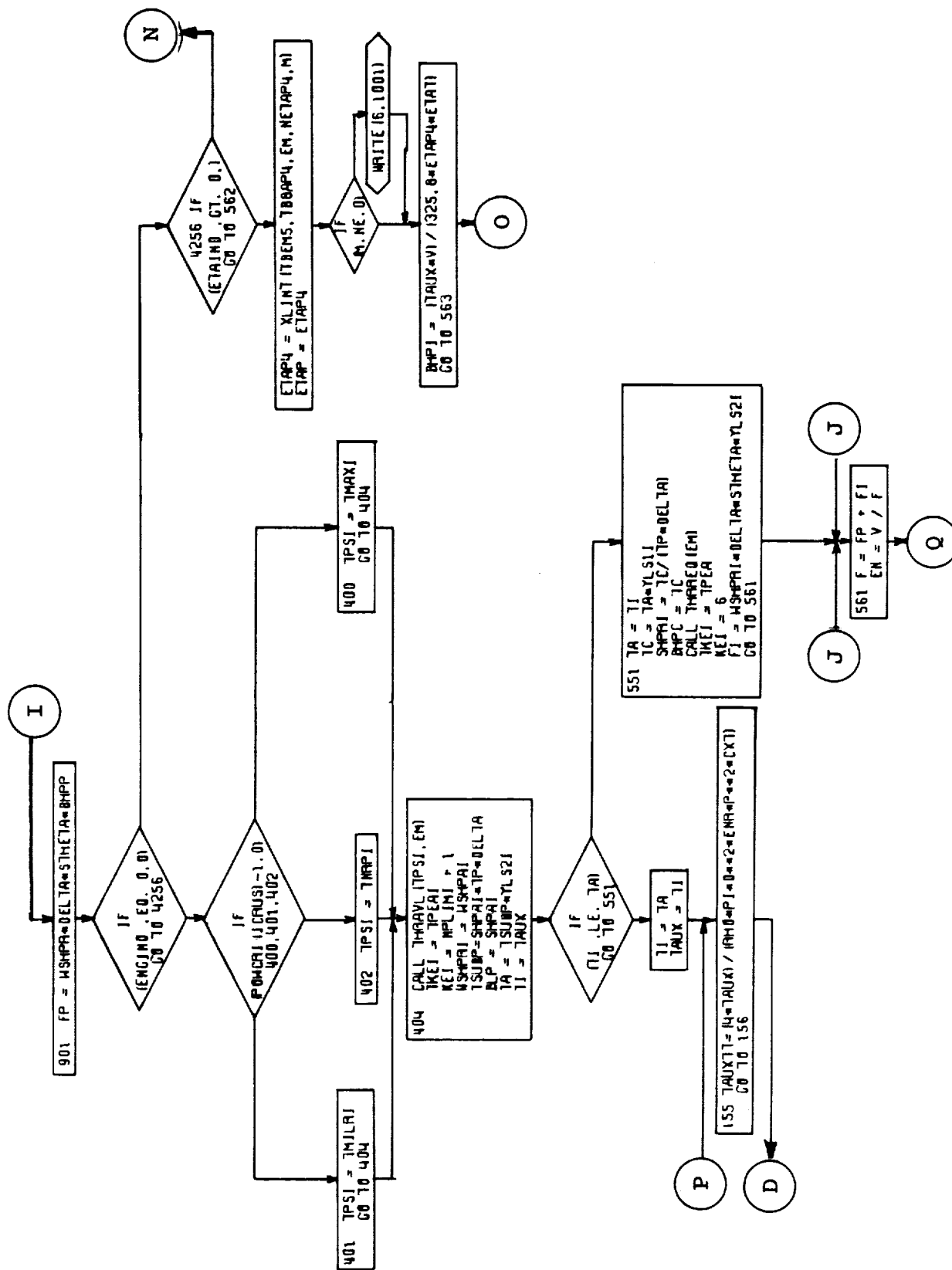


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 5 of 8).

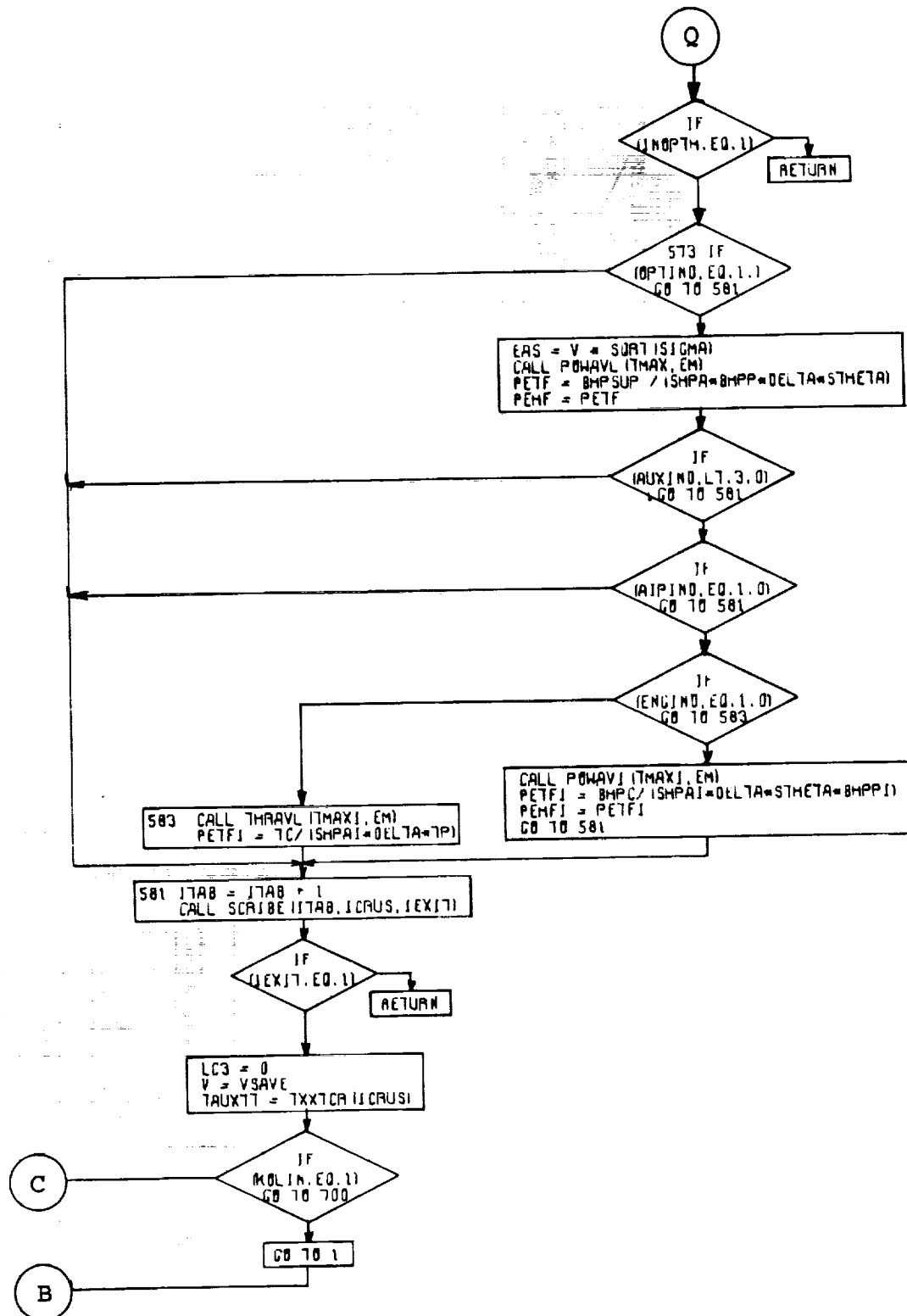


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 6 of 8).

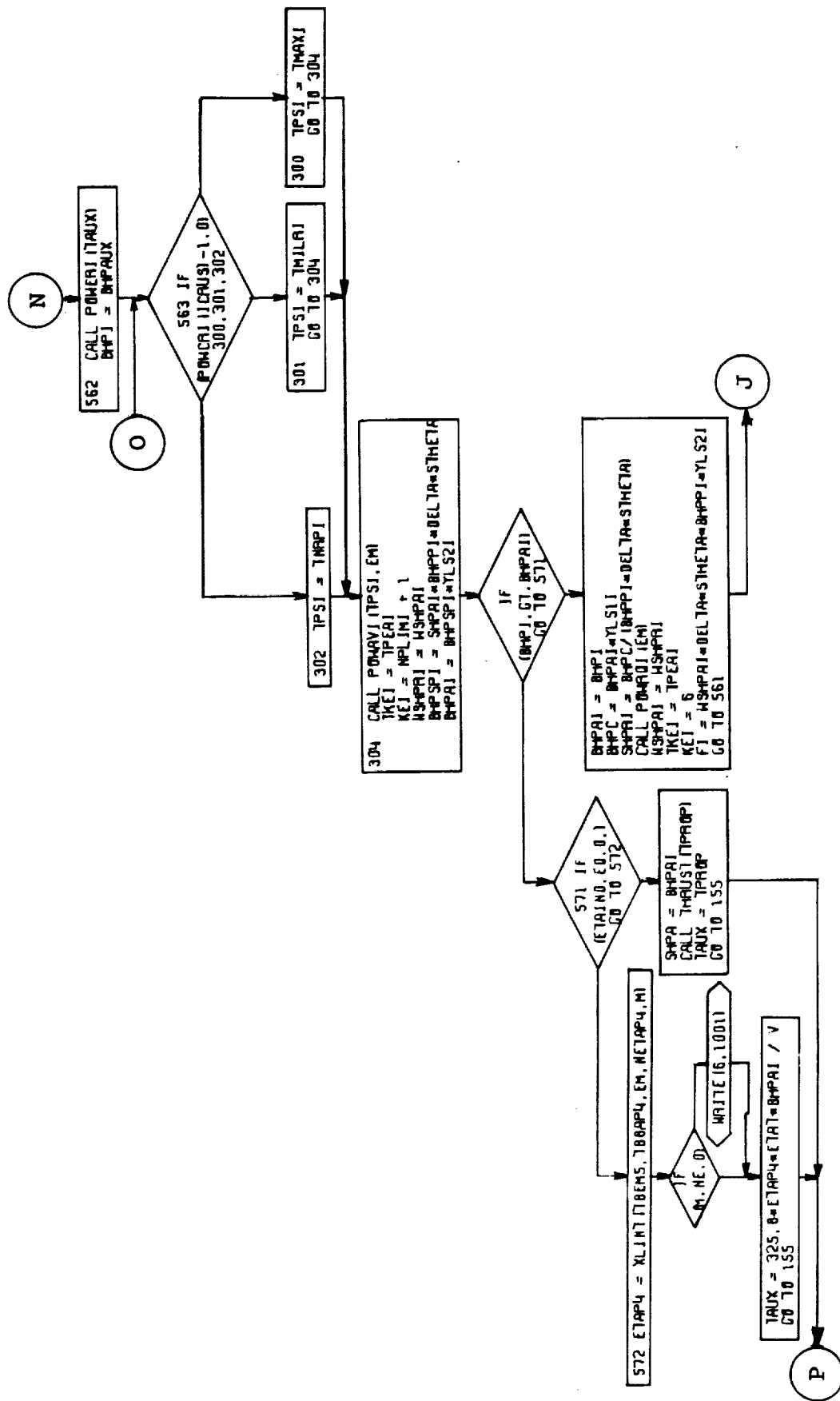


Figure 4-45b. Cruise Calculations Subroutine Flow Chart (Part 7 of 8).

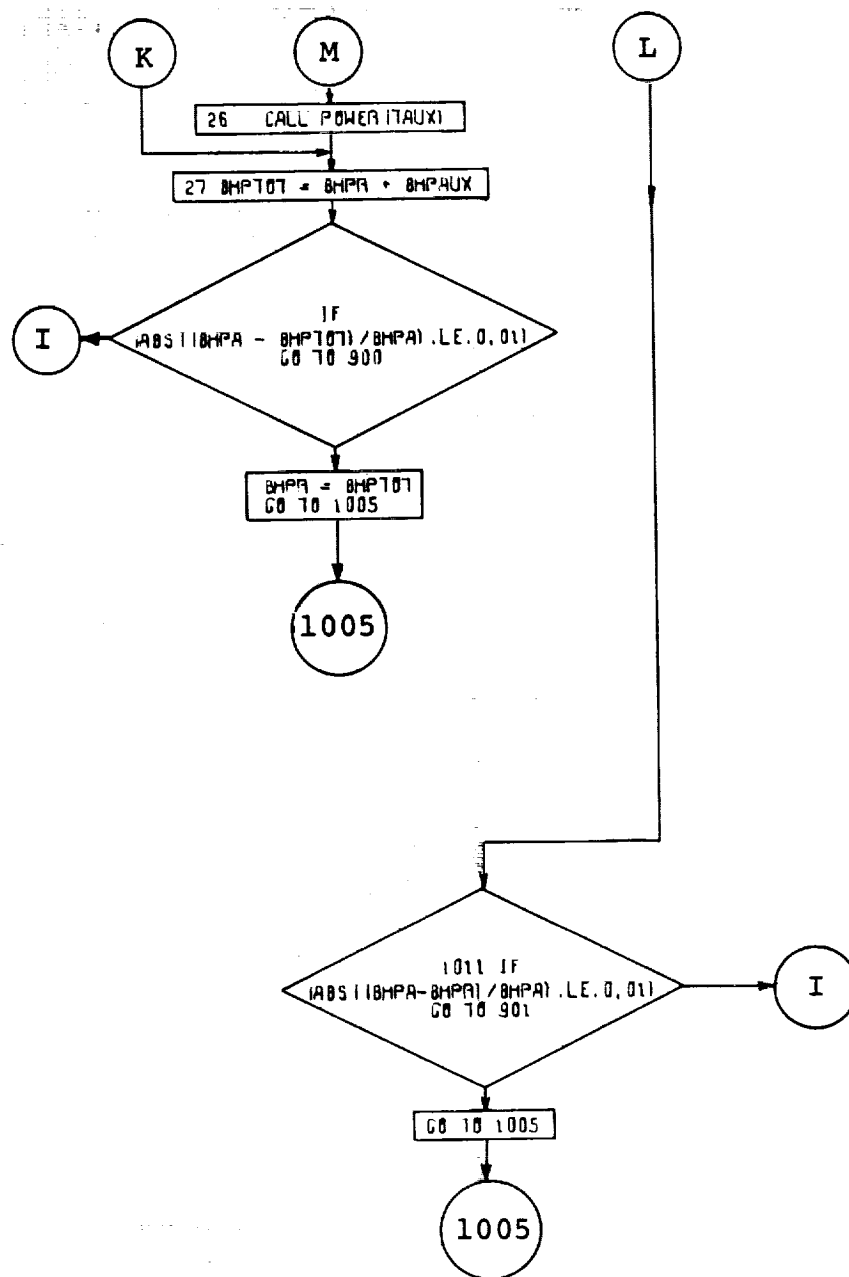


Figure 4-45b. Cruise Calculations Subroutine Flow Chart
(Part 8 of 8).

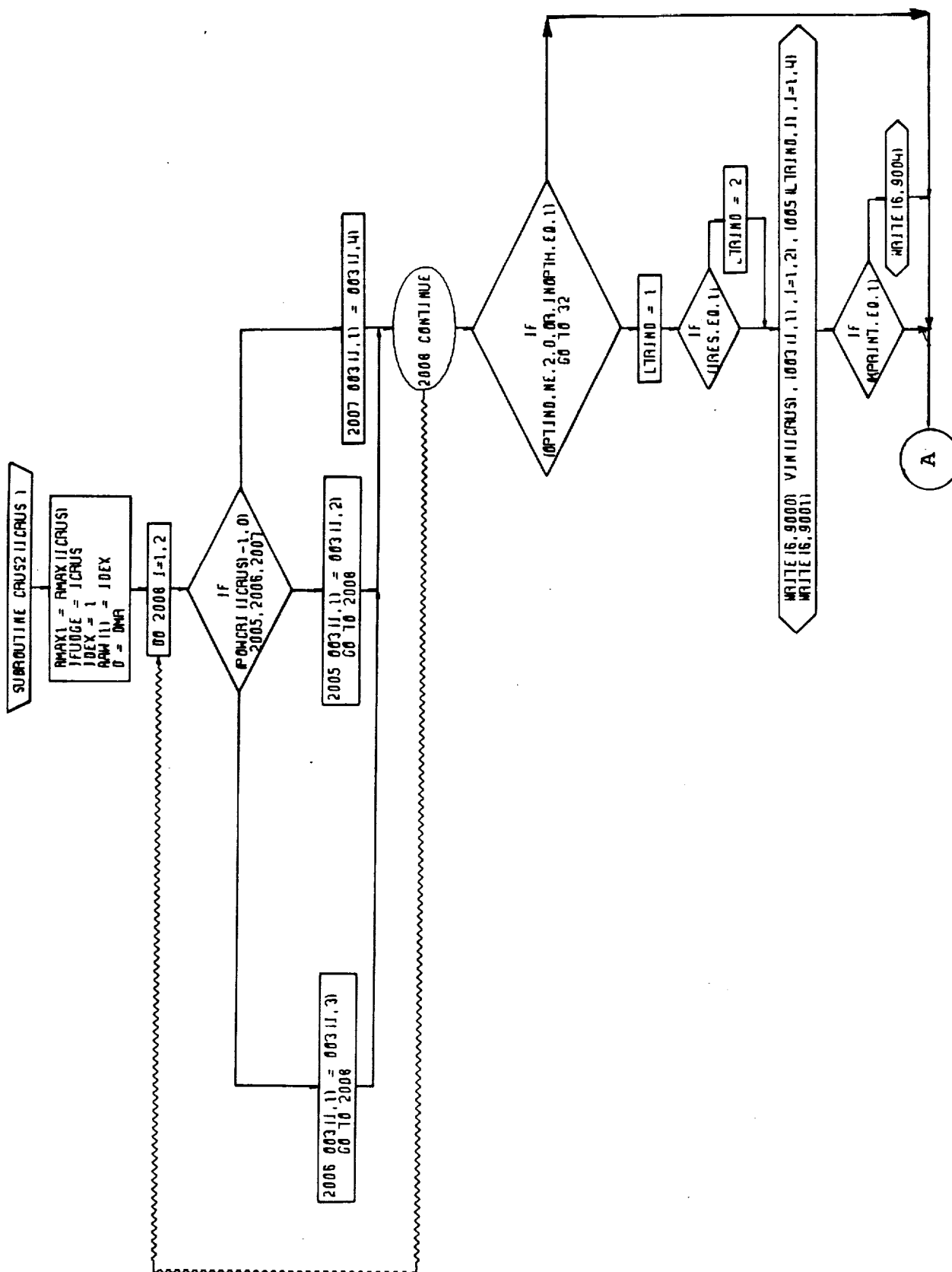


Figure 4-45c. Cruise Calculations Subroutine Flow Chart (Part 1 of 7).

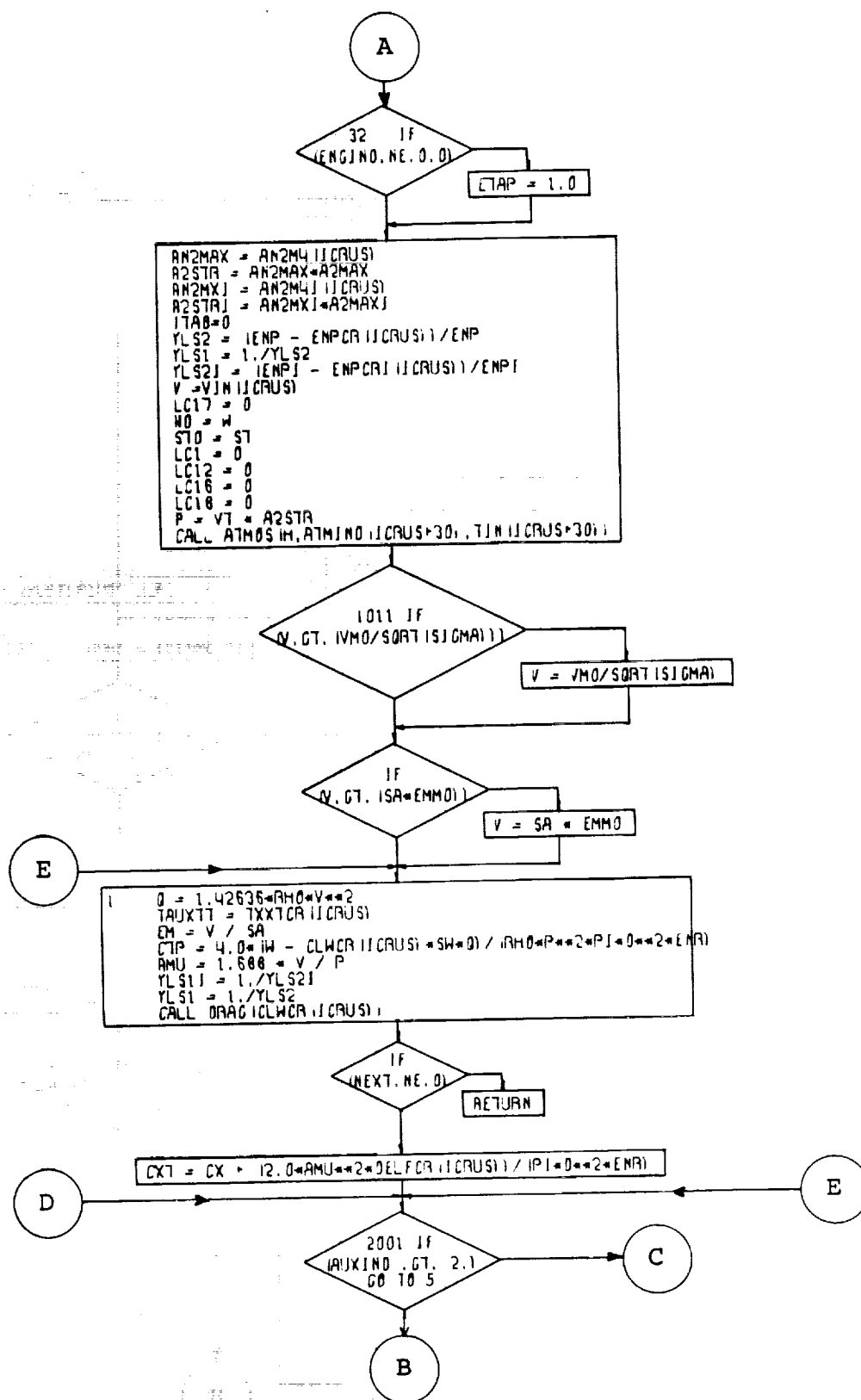


Figure 4-45c. Cruise Calculations Subroutine Flow Chart
(Part 2 of 7).

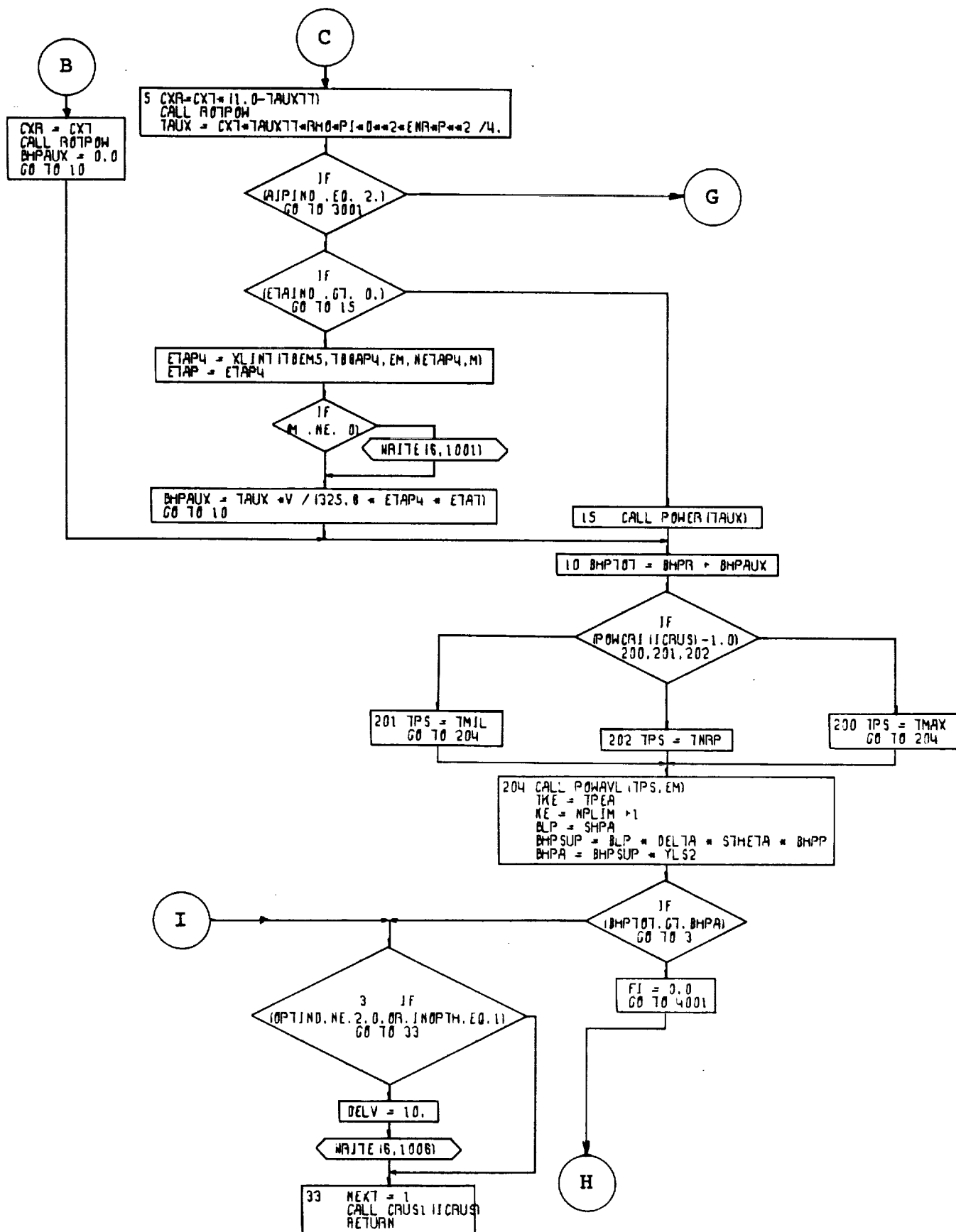


Figure 4-45c. Cruise Calculations Subroutine Flow Chart (Part 3 of 7).

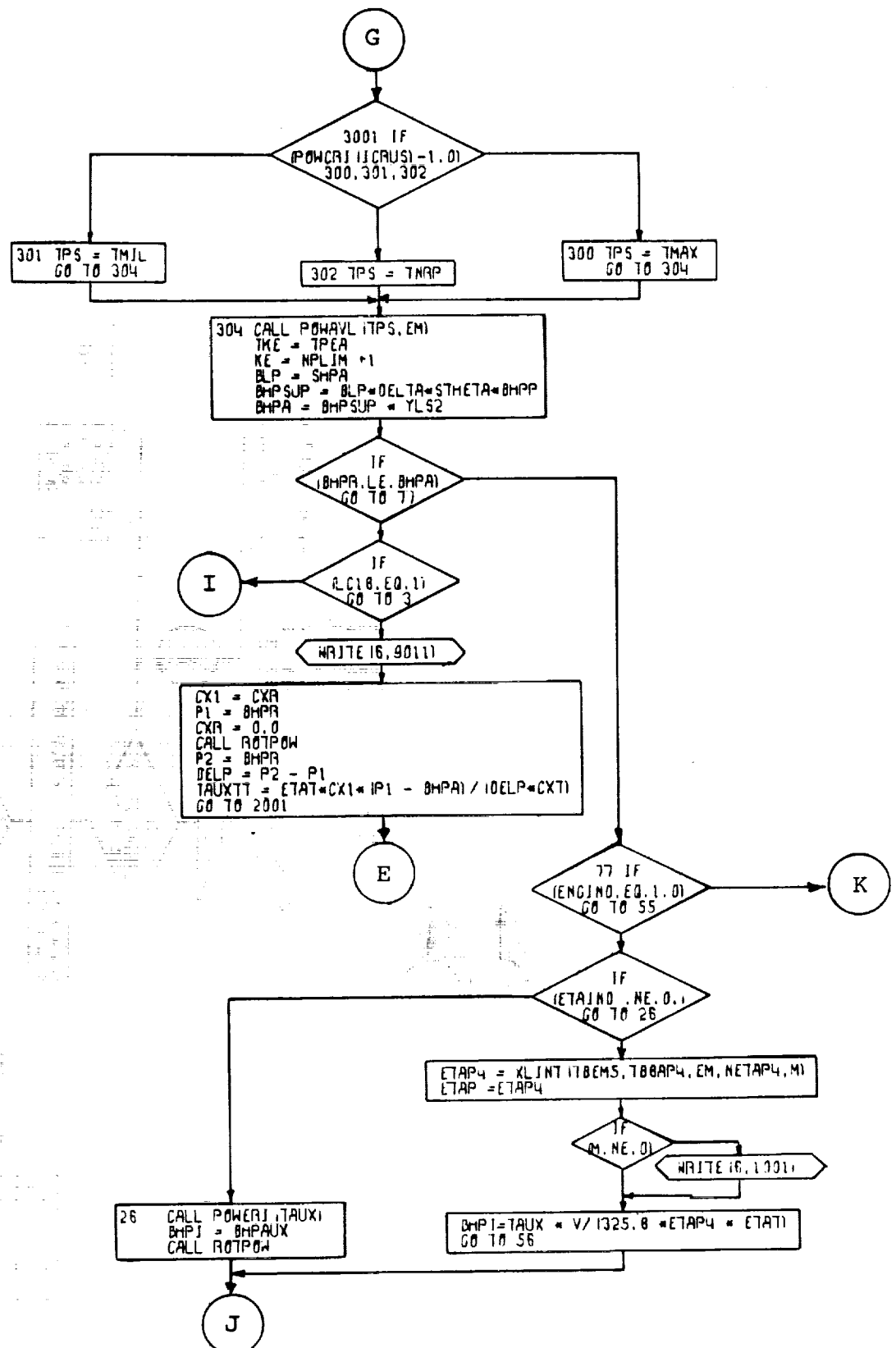


Figure 4-45c. Cruise Calculations Subroutine Flow Chart (Part 4 of 7).

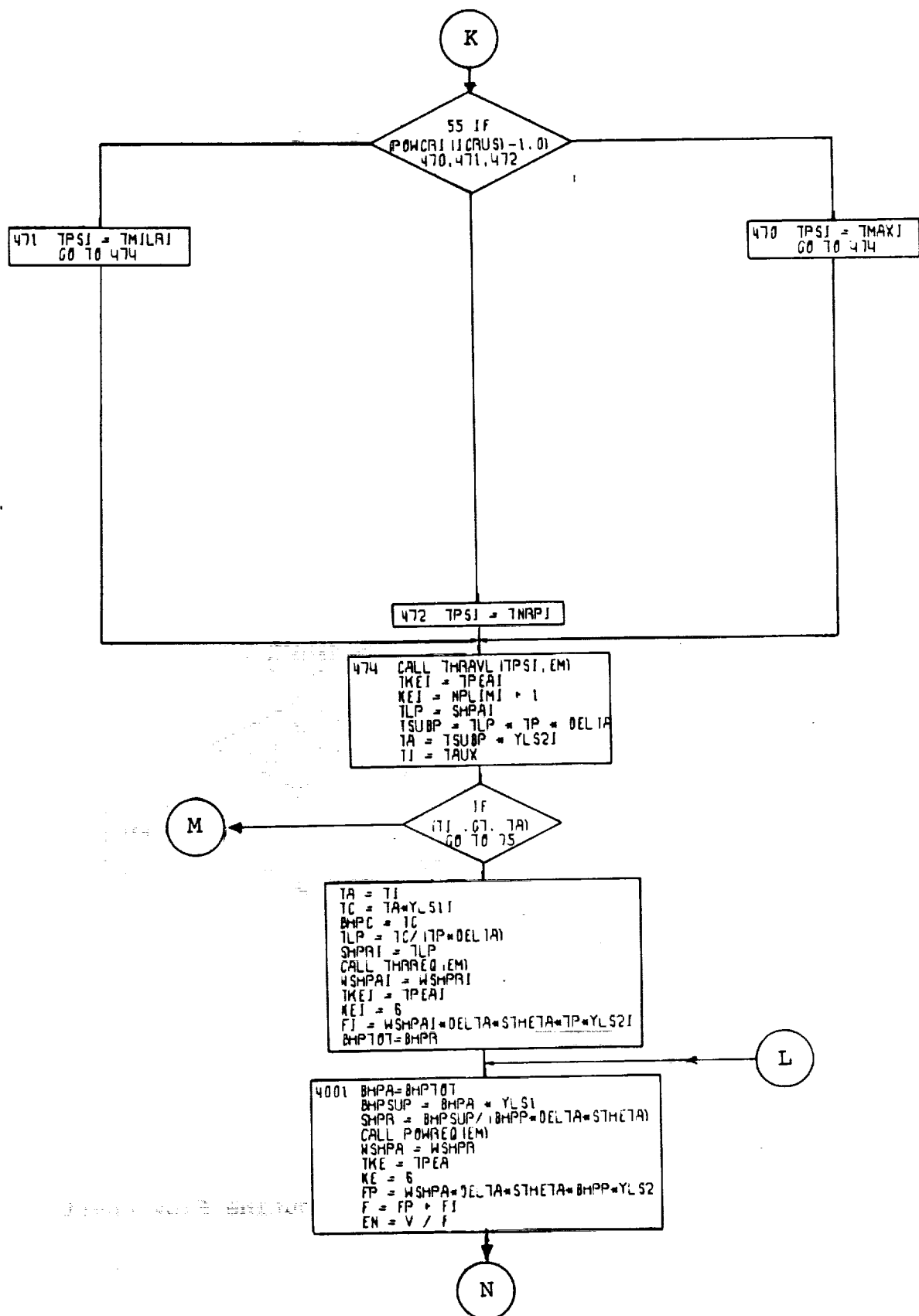


Figure 4-45c. Cruise Calculations Subroutine Flow Chart
(Part 6 of 7).

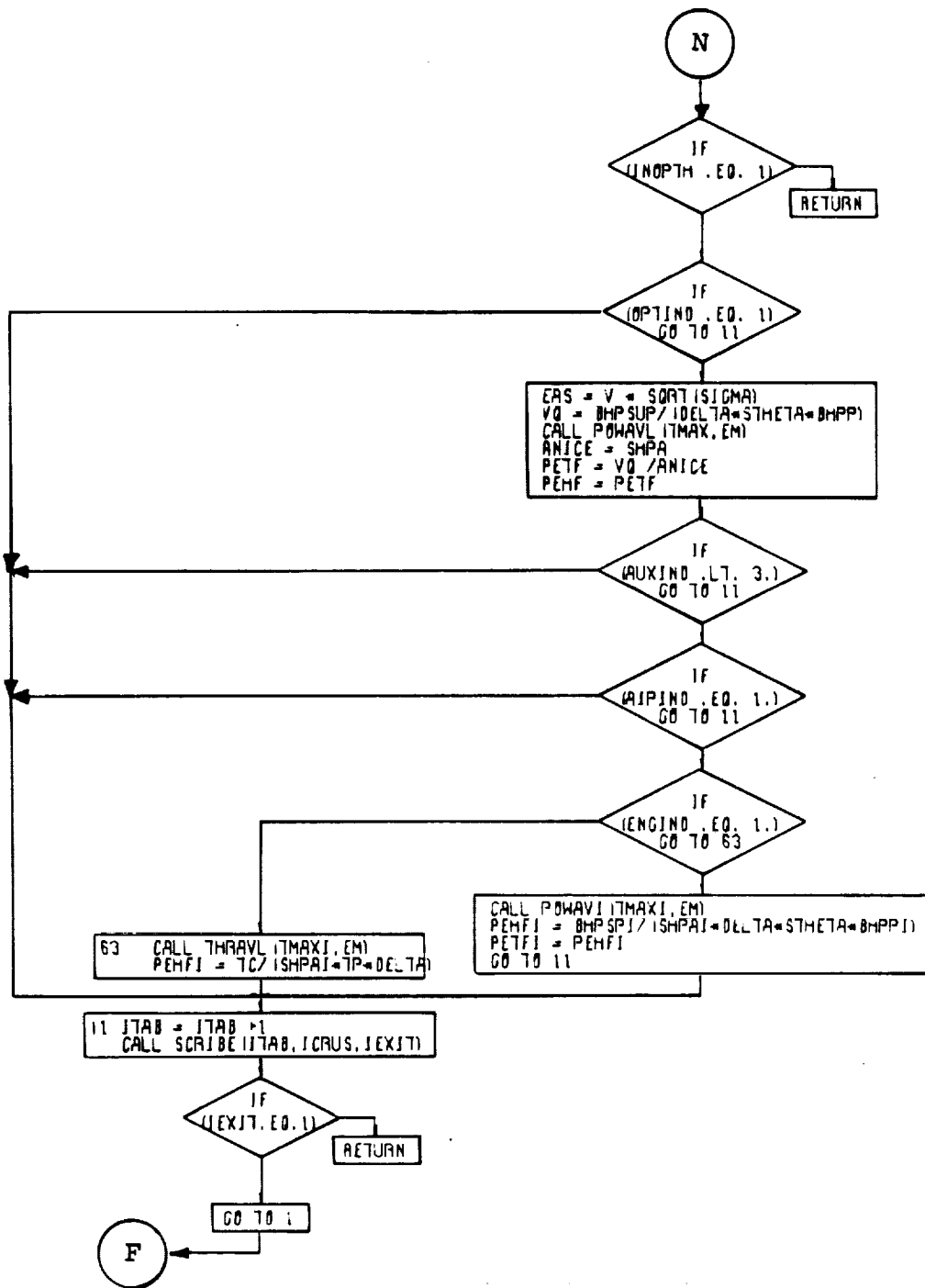


Figure 4-45c. Cruise Calculations Subroutine Flow Chart
(Part 7 of 7).

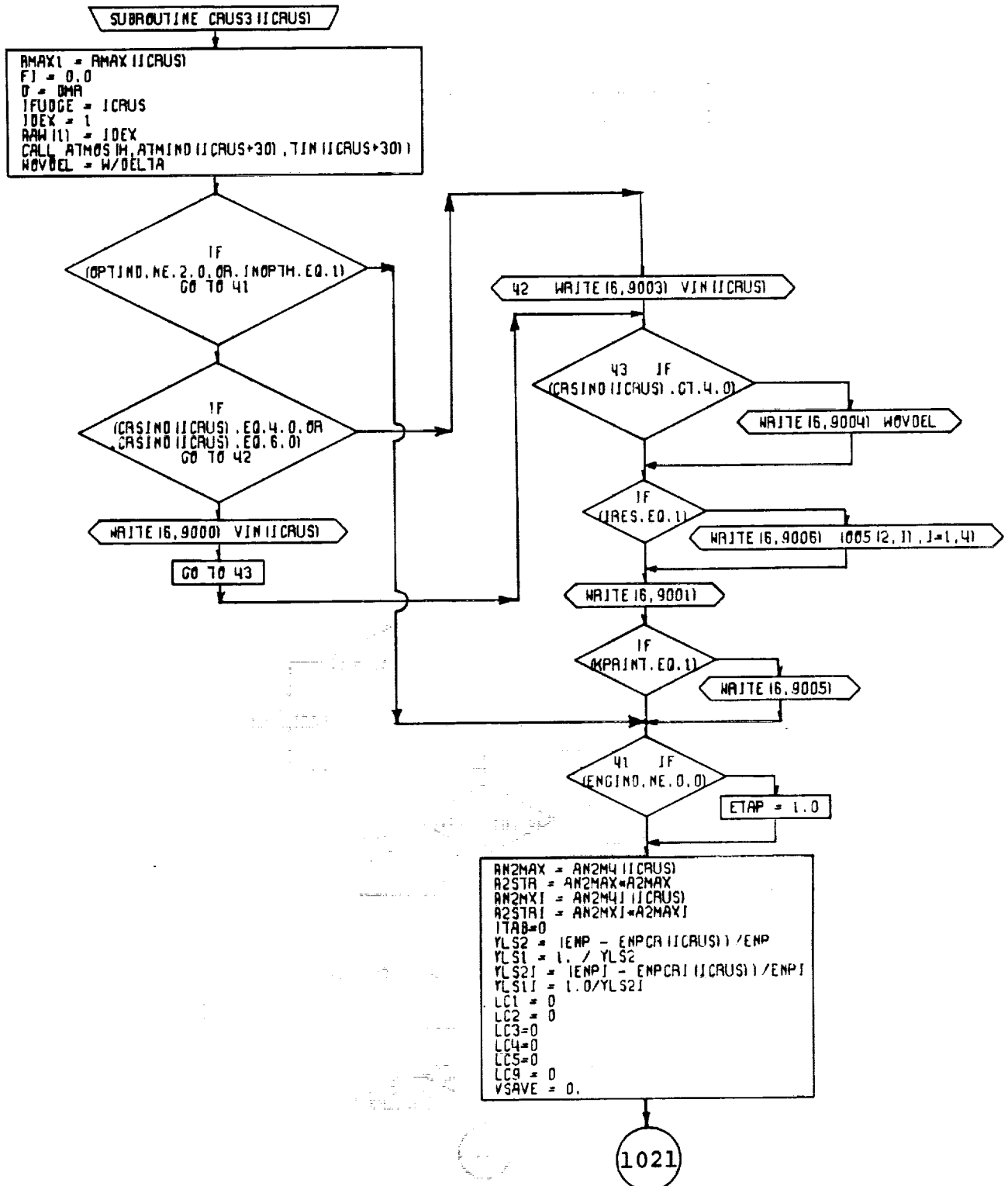


Figure 4-45d. Cruise Calculations Subroutine Flow Chart
(Part 1 of 6).

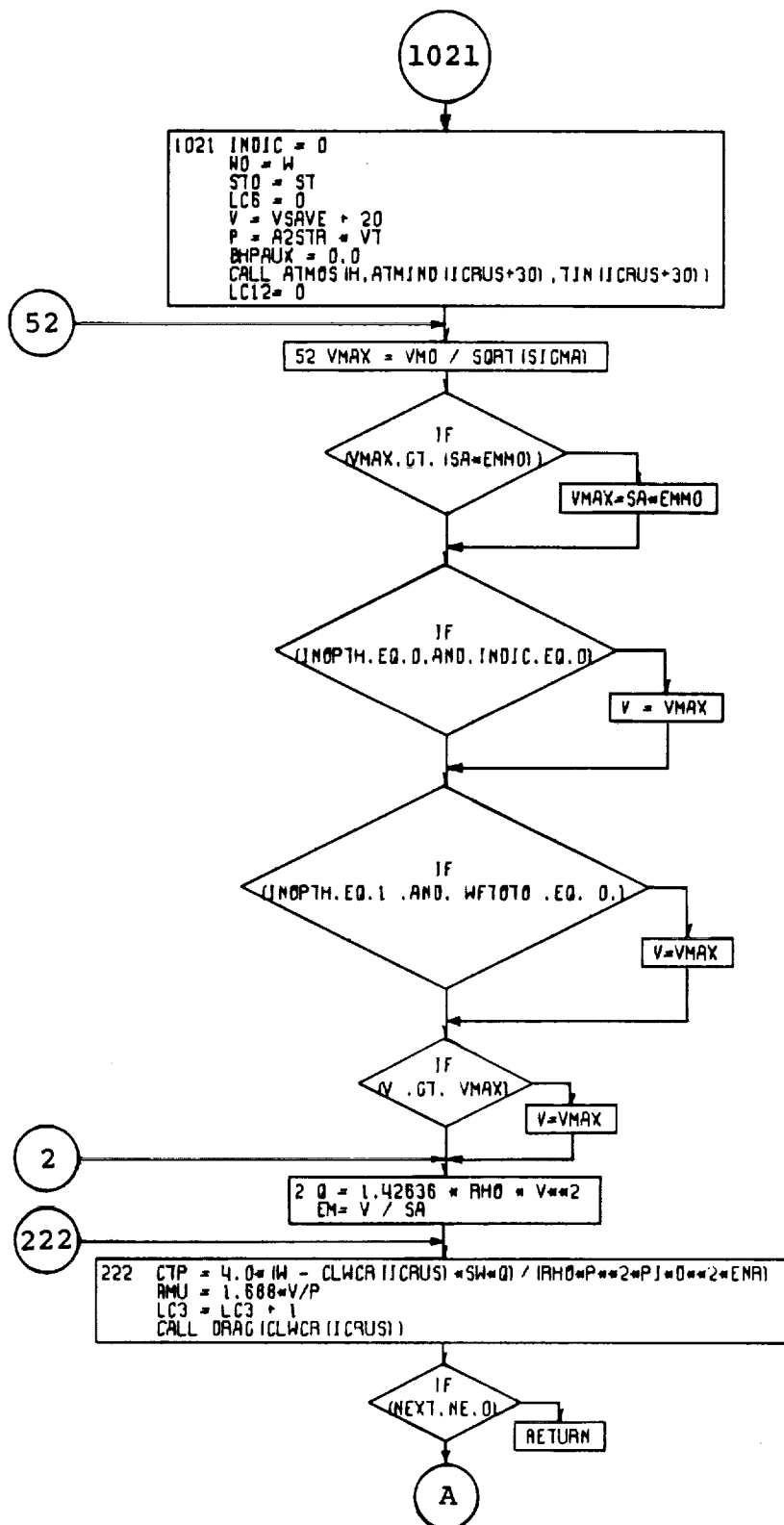


Figure 4-45d. Cruise Calculations Subroutine Flow Chart (Part 2 of 6).

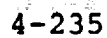


Figure 4-45d. Cruise Calculations Subroutine Flow Chart (Part 3 of 6).

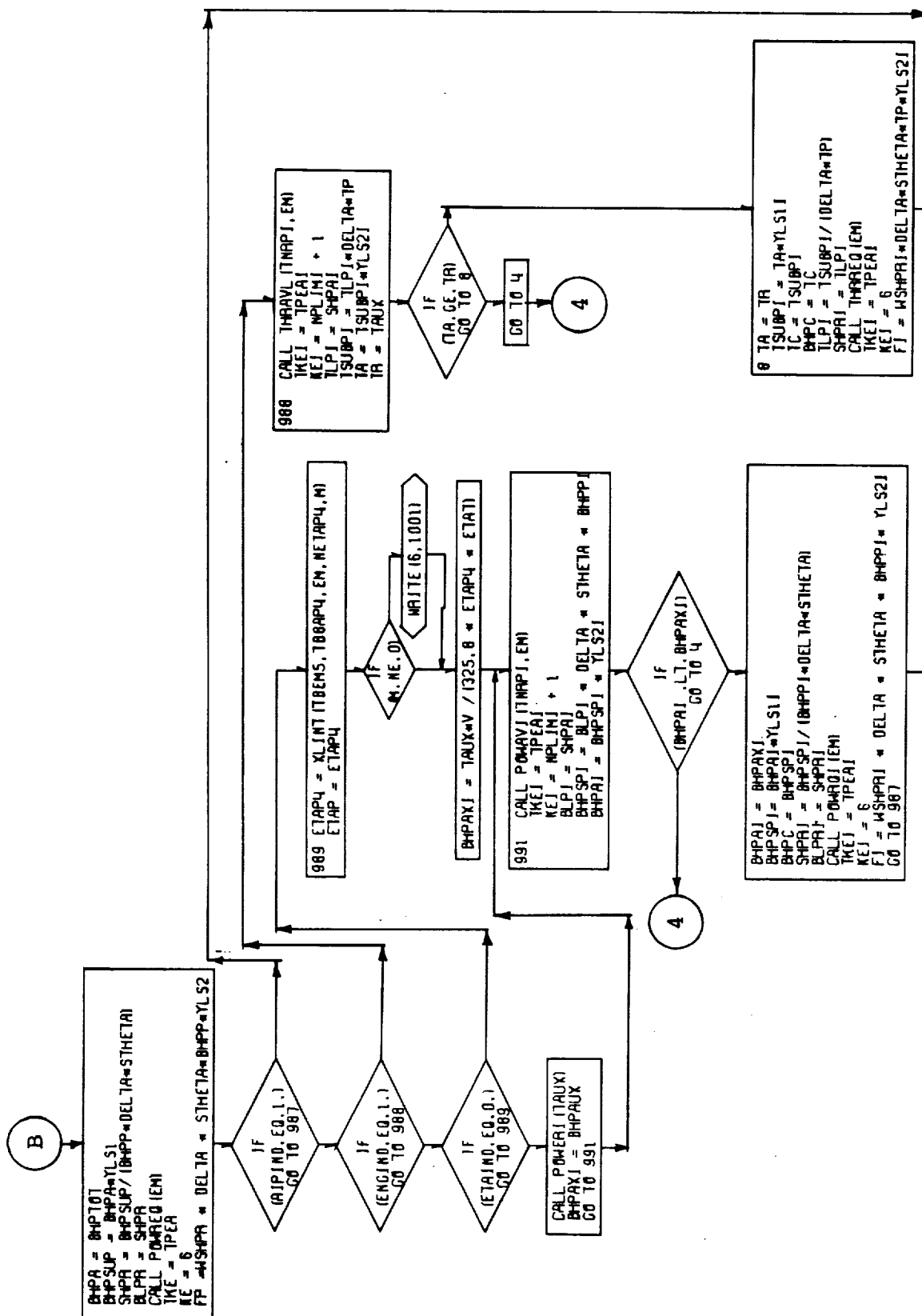
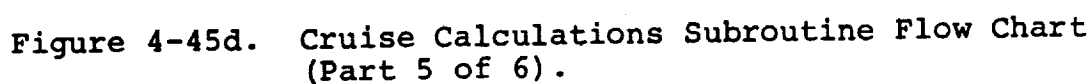


Figure 4-45d. Cruise Calculations Subroutine Flow Chart (Part 4 of 6).



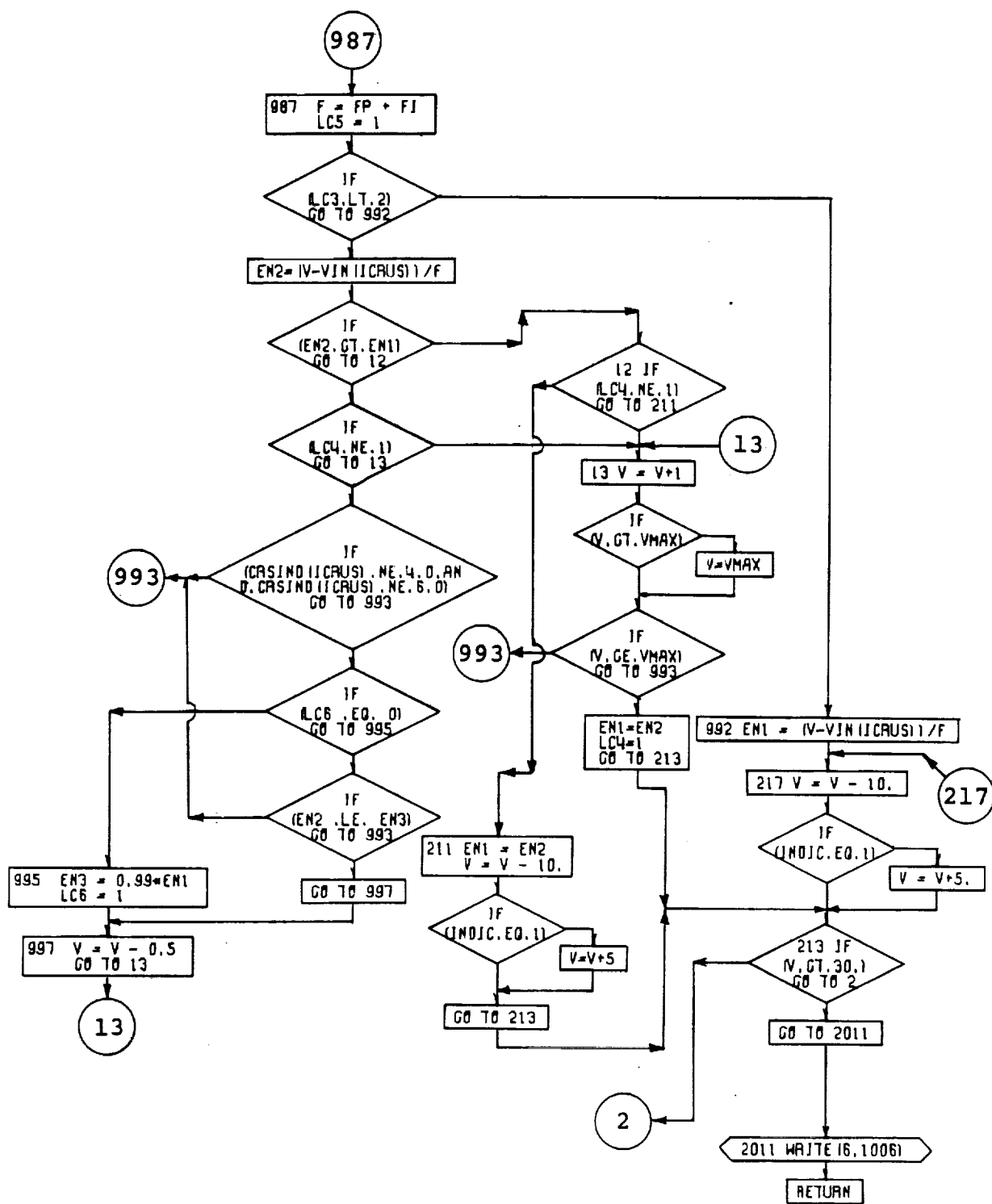


Figure 4-45d. Cruise Calculations Subroutine Flow Chart (Part 6 of 6).

4.12.5 Descent Calculations Subroutine

Twelve different options for descent performance calculation are available. The options fall into three different categories: descent at constant true air speed (TAS), descent at constant equivalent air speed (EAS), and descent at constant Mach number. In addition, each type of descent may be calculated for a specified type of descent flight path, specified by RMAXND as follows:

<u>Value of RMAXND</u>	<u>Type of Flight Path</u>
0	Descent flight path ends at specified terminal range.
1	Program checks terminal range requirement, but does not match it.
2	Descent flight path ends at a specified minimum altitude, the terminal range requirement not being considered.
3	Fuel used and time required for descent are calculated, but no range credit is given (i.e., spiral descent path).

Rate of descent (R/D) is always input. If the airspeed-rate of descent combination (as specified by DESIND) exceeds the descent boundaries, the airspeed will be held constant, while the R/D is adjusted accordingly. Figure 4-46 in which R/D is plotted against flight speed, illustrates the descent boundaries. They are:

- (a) Vertical rate of descent boundary

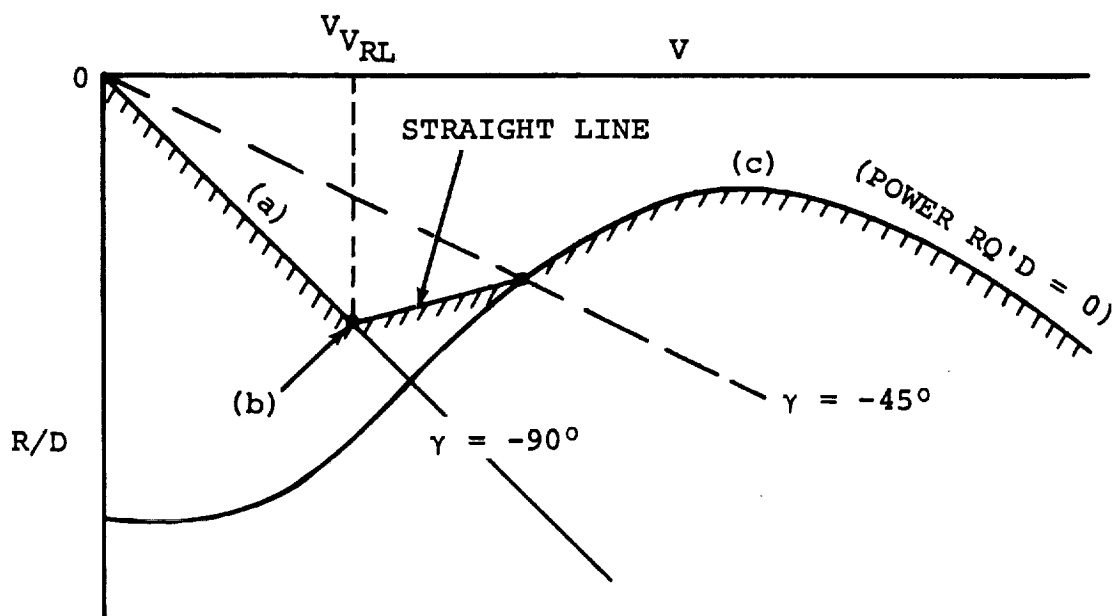
$$(\gamma = -90^\circ)$$

- (b) V_{VRL} (vortex ring state) limit descent speed - defined by the equation

$$V_{VRL} = \frac{2}{3} \sqrt{\frac{T}{2\rho A}}$$

where T = total rotor thrust
 A = disc area.

- (c) Autorotative descent (power required = 0).



- (a) Vertical rate of descent boundary
- (b) Vortex ring state limit descent speed
- (c) Autorotative descent boundary

Figure 4-46. Descent Boundaries.

The distinction between the first type of descent flight path (as specified by RMAXND = 0) and the last three (RMAXND = 1, 2, 3) in regard to the range at which the descent starts, when terminal range is specified, should be clearly understood.

- a. If RMAXND = 0, no spiral descent path is permitted. The program will calculate the value for range at the beginning of the descent which is required to satisfy the terminal condition on range and altitude. In order to do this, the program "backs up" on the previous segment. If this option (RMAXND = 0) is used, the descent must be preceded by a cruise segment. The input value for maximum range for the preceding cruise segment is a dummy value and the cruise will actually terminate, in order to begin descent, at an earlier point. It is recommended, however, that when the RMAXND = 0 option is to be used, the maximum range during the preceding cruise be input as the same value as the terminal range at the end of the following descent.
- b. If RMAXND = 1, the descent will start at the current value for range and, as previously described, the aircraft will fly a straight-line path to the desired terminal point. If the predicted flight path (as checked against the specified terminal range by the program) ends beyond the specified terminal range, a spiral descent path from the altitude at that point to the final altitude is assumed. If the predicted flight path ends before reaching the specified terminal range point, the program prints "SHALLOWER DESCENT REQUIRED". The descent may follow any other segment (climb, cruise, etc.) or may start the mission.
- c. If RMAXND = 2 or 3, the descent will start at the current value for range; and, depending on the option chosen, will either end at the minimum altitude (RMAXND = 2) specified, with no constraint on the resulting range, or a spiral descent path (RMAXND = 3) will be assumed. As with RMAXND = 1, the descent options may follow any other segment, or may start the mission.

An increment in aircraft parasite drag may be input in order to simulate the effects of dive brakes, external stores, etc. It is possible to use a descent segment in the mission profile to account for a reserve fuel requirement (SGTIND = 50) (in such a case the helicopter weight at the end of descent is set back to the weight at the beginning of descent) or as a part of the basic mission (in this case the weight is not reset). In either case, the fuel used during descent is included in the total fuel required to size the helicopter.

Input to the subroutine consists of the settings for DESIND

and RMAXND, atmospheric conditions, R/D, the propulsive thrust split, the incremental parasite drag, the operating wing lift coefficient (winged and compound helicopters), the final altitude, the step size, the required TAS, EAS or Mach number, the terminal range requirement (RMAXND = 0, 1) and the number of engines shut down. Subroutine DESPOW (which is called by the Descent calculations subroutine) calculates the power required for descent at the desired flight conditions. Figure 4-47 is a flow chart of subroutine DESPOW and Figure 4-48 is a flow chart of the descent calculations subroutine.

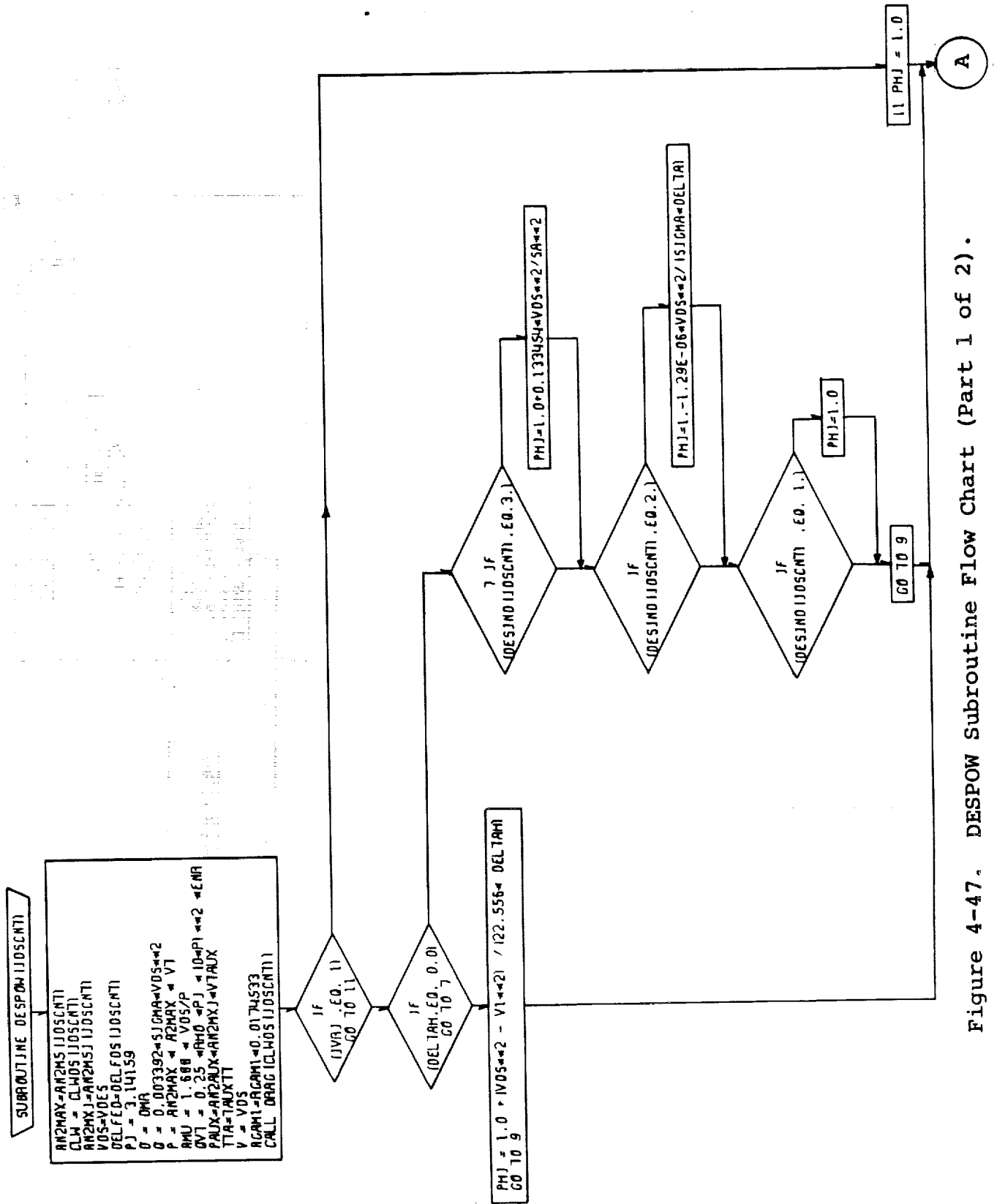


Figure 4-47. DESPOW Subroutine Flow Chart (Part 1 of 2).

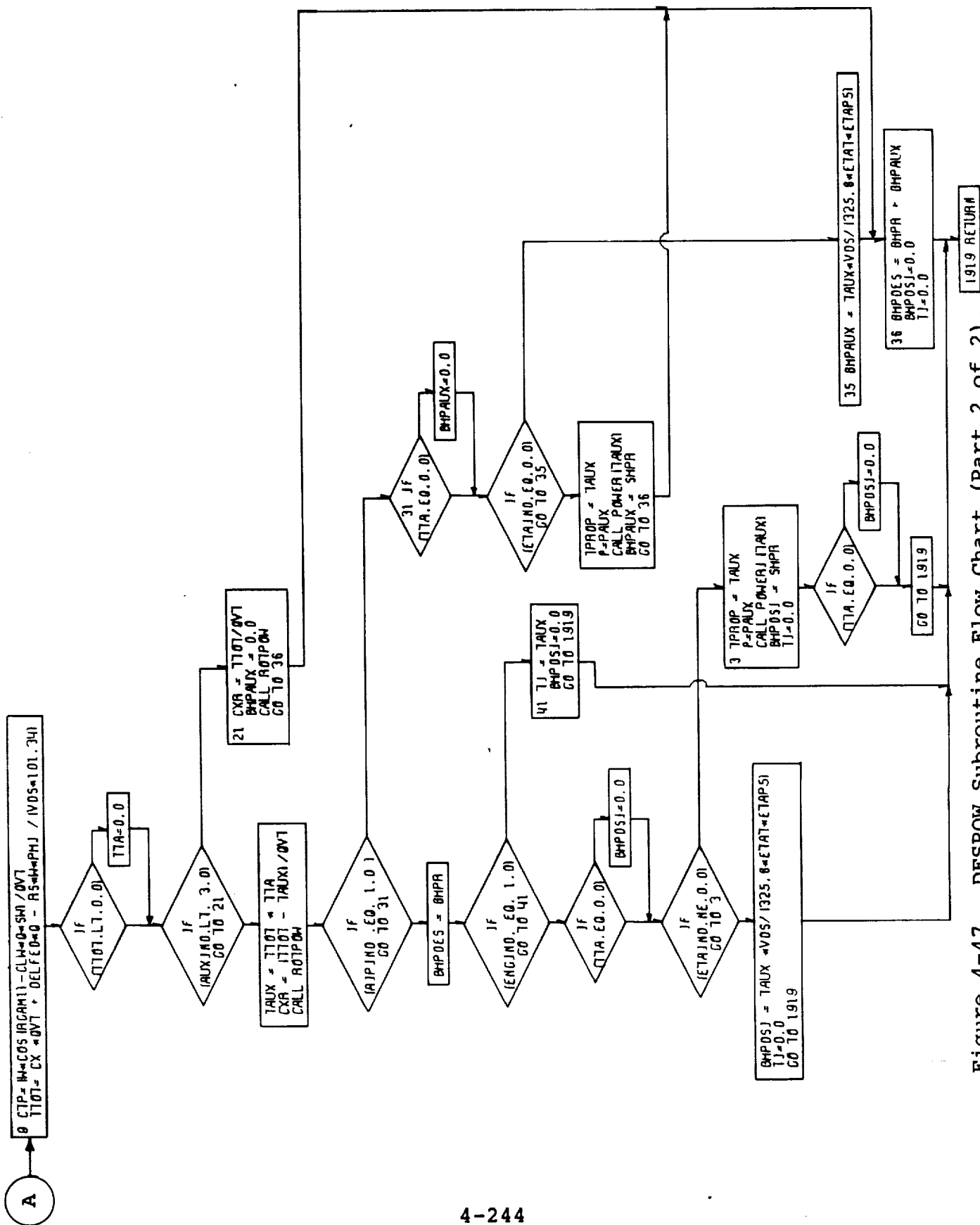


Figure 4-47. DESPOW Subroutine Flow Chart (Part 2 of 2).

SUBROUTINE DESCENT

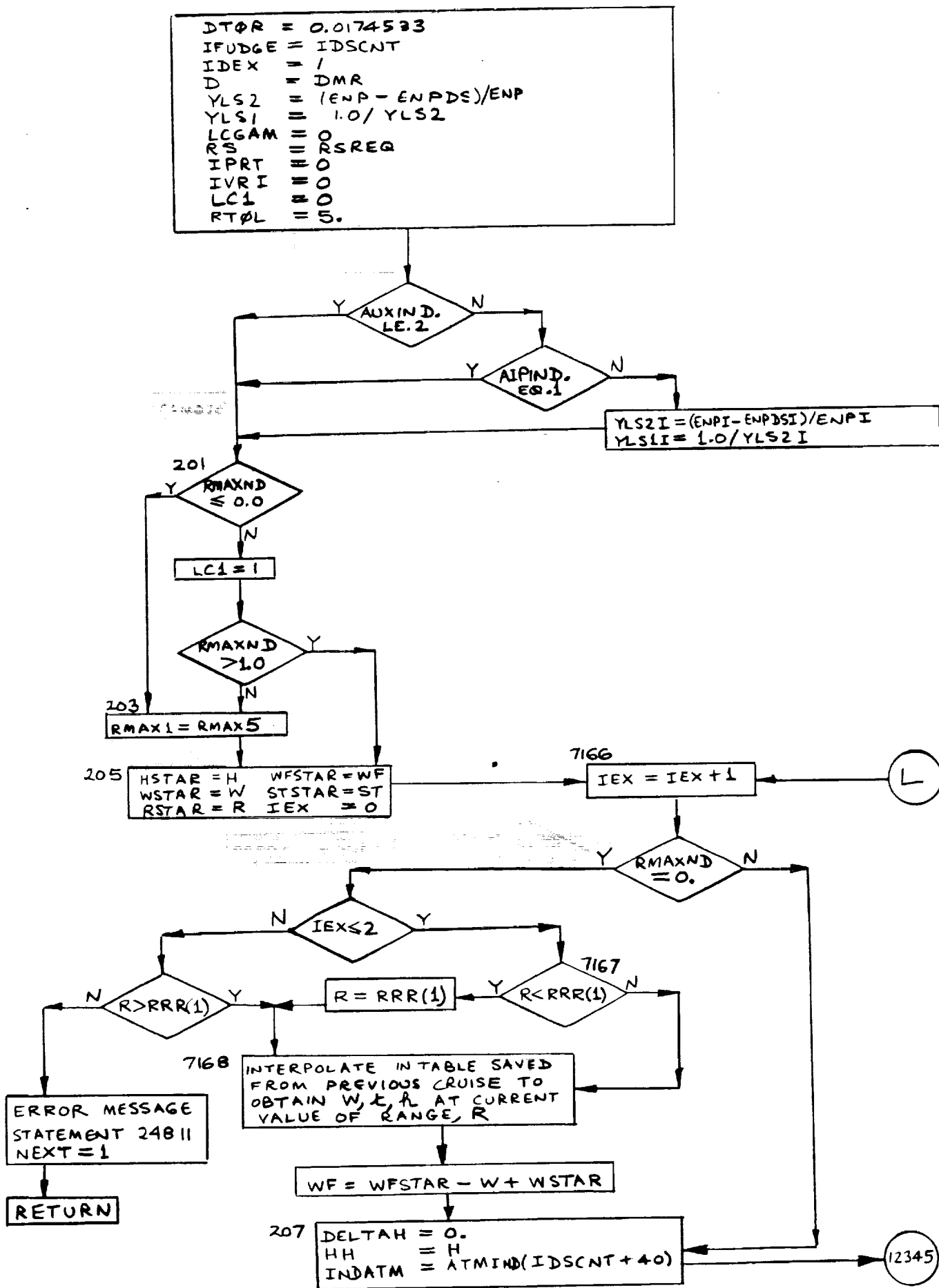


Figure 4-48. Descent Calculations Subroutine Flow Chart (Part 1 of 6).

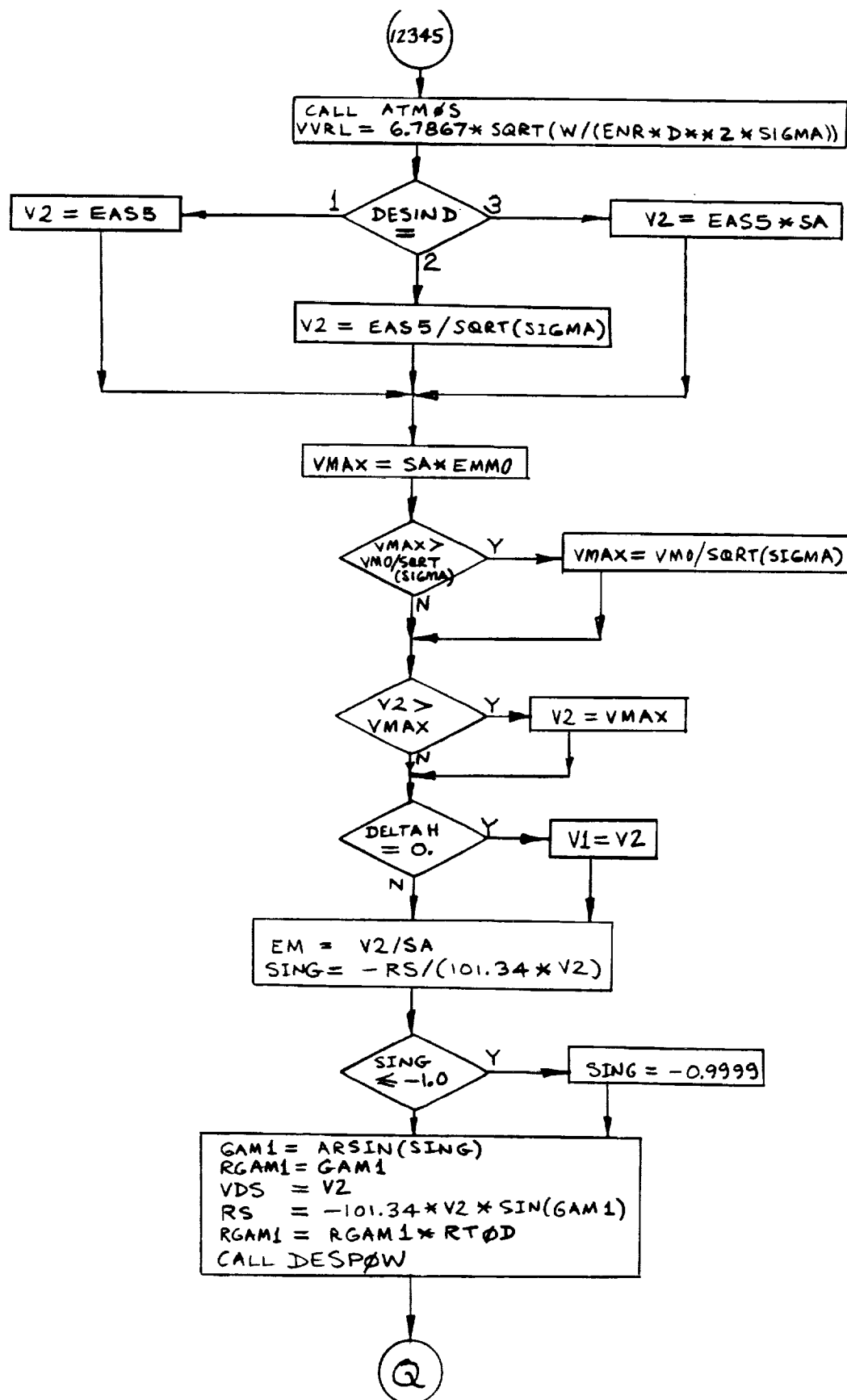


Figure 4-48. Descent Calculations Subroutine Flow Chart (Part 2 of 6).

Q

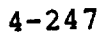
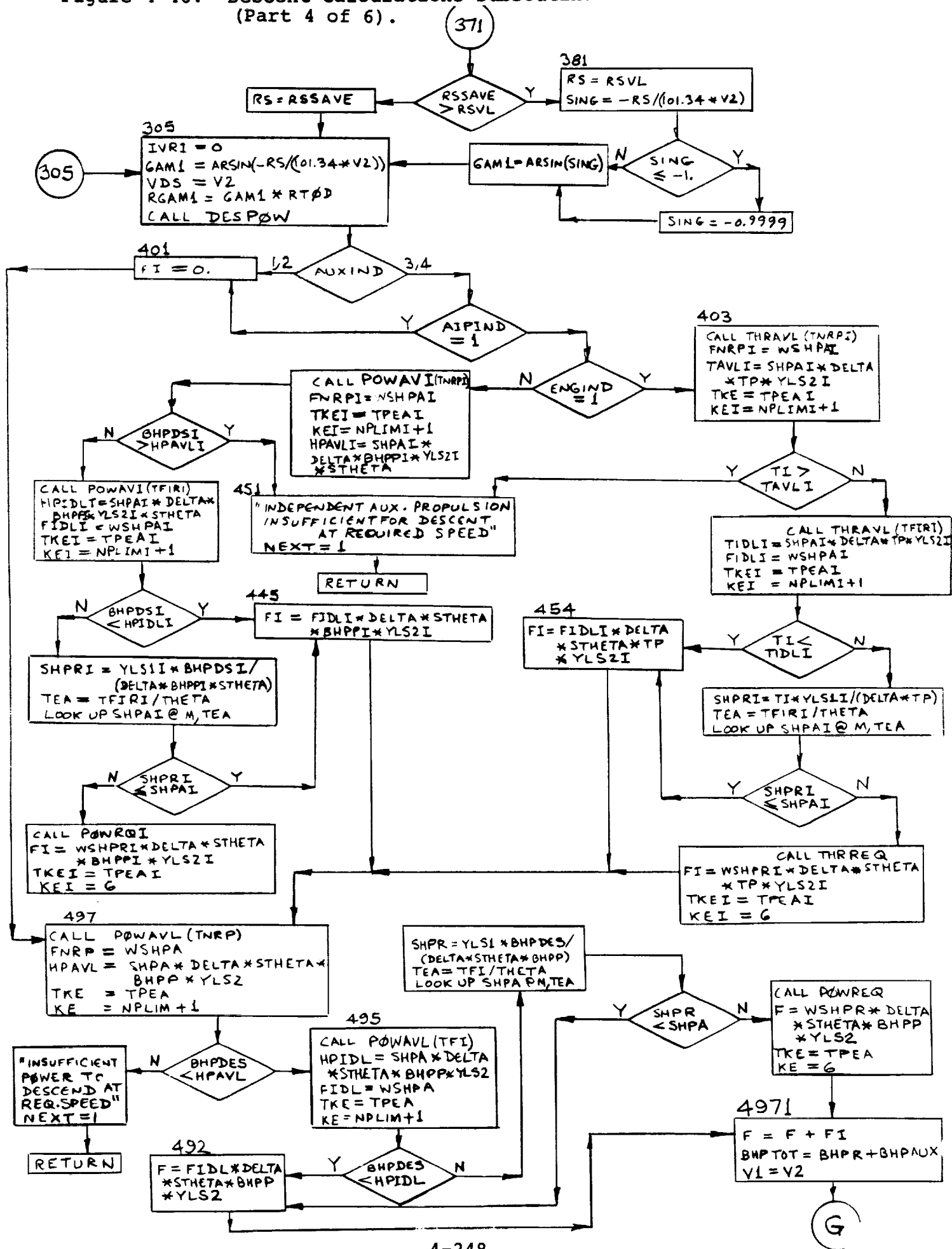
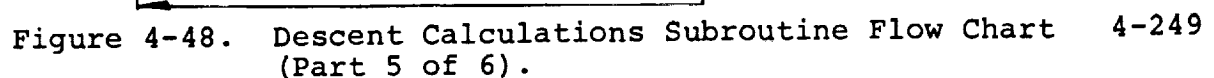


Figure 4-48. Descent Calculations Subroutine Flow Chart
(Part 4 of 6).





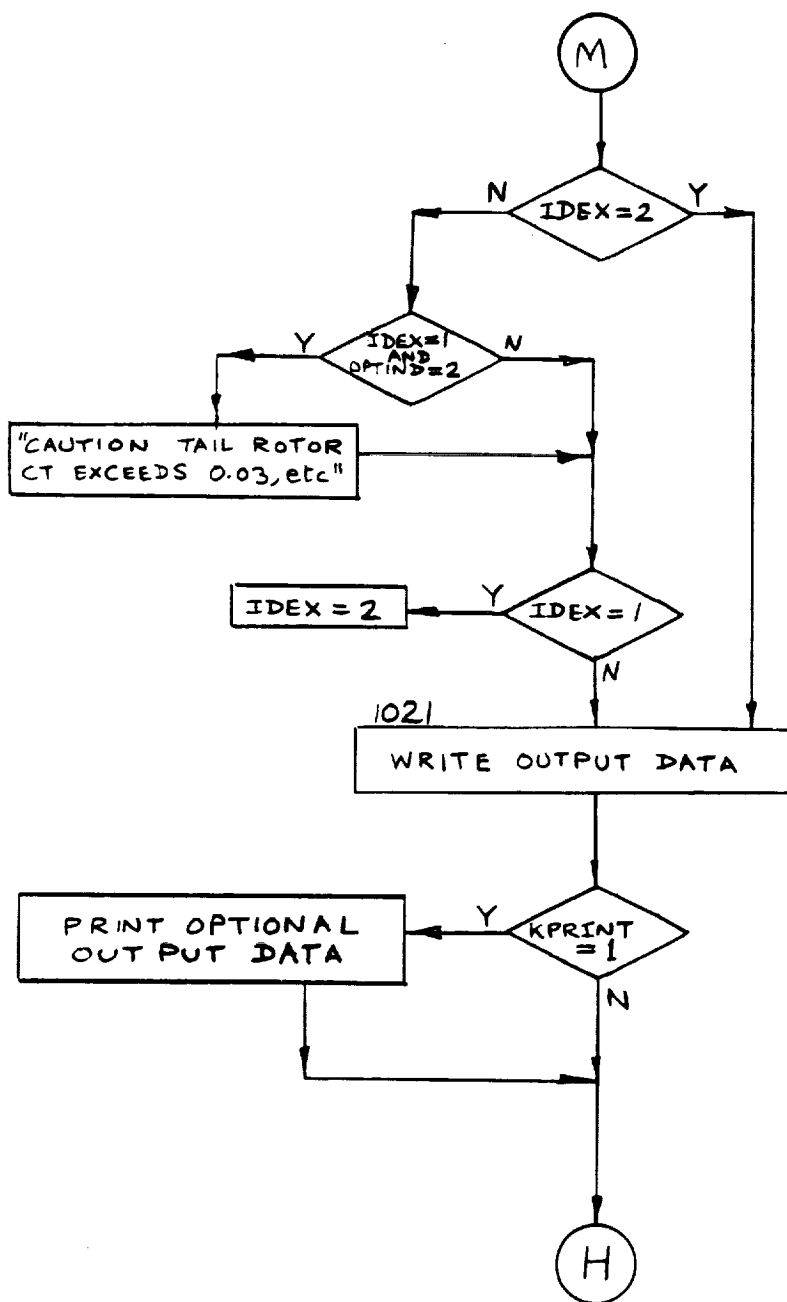


Figure 4-48. Descent Calculations Subroutine Flow Chart
(Part 6 of 6).

4.12.6 Loiter Calculations Subroutine

The sixth performance segment represents a calculation of helicopter loiter performance. In this subroutine, the helicopter will fly at the airspeed for best endurance. This subroutine calculates the power required and the airspeed to maximize the endurance of the helicopter. It also determines the fuel required to loiter for a specified period of time.

Engine shutdown during loiter may be simulated by inputs for Npsd (primary engines) and Npsdi (auxiliary independent engines). One or more engines may be shutdown. An increment in helicopter drag ($\Delta F_{eLOITER}$) may be input to represent drag changes due to external stores, windmilling propellers (in the case of a compound helicopter using propellers), etc.

For a compound helicopter, the split of propulsive thrust required between the main rotor and the auxiliary propulsion system may be specified by an input for (T_{AUX}/T_{TOT}) .

It is possible to use a loiter segment in the mission profile to account for a reserve fuel requirement ($SGTIND = 60$) (in such case the helicopter weight at the end of loiter is set back to the weight at the beginning of loiter) or as a part of the basic mission (in this case the weight is not reset). In either case, the fuel used during loiter is included in the total fuel required to size the helicopter.

The input to this subroutine consists of the time for loiter, step size (incremental time), the incremental parasite drag area, the number of engines (primary and auxiliary independent) shut down, the atmospheric conditions, the operating wing lift coefficient (in the case of compound and winged helicopters), and the propulsive thrust split. A flow chart of this subroutine is shown in Figure 4-49.

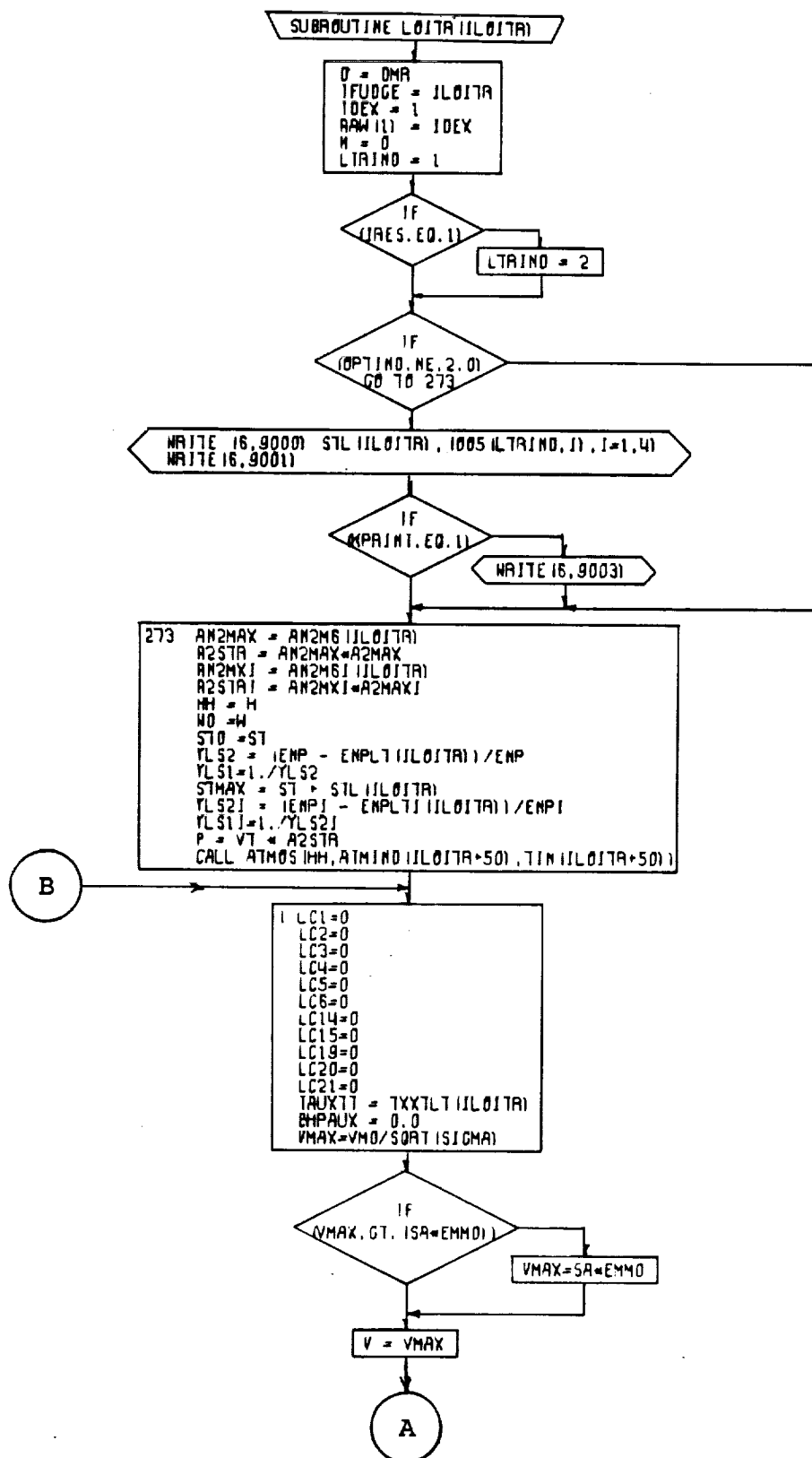


Figure 4-49. Loiter Calculations Subroutine (Part 1 of 7).

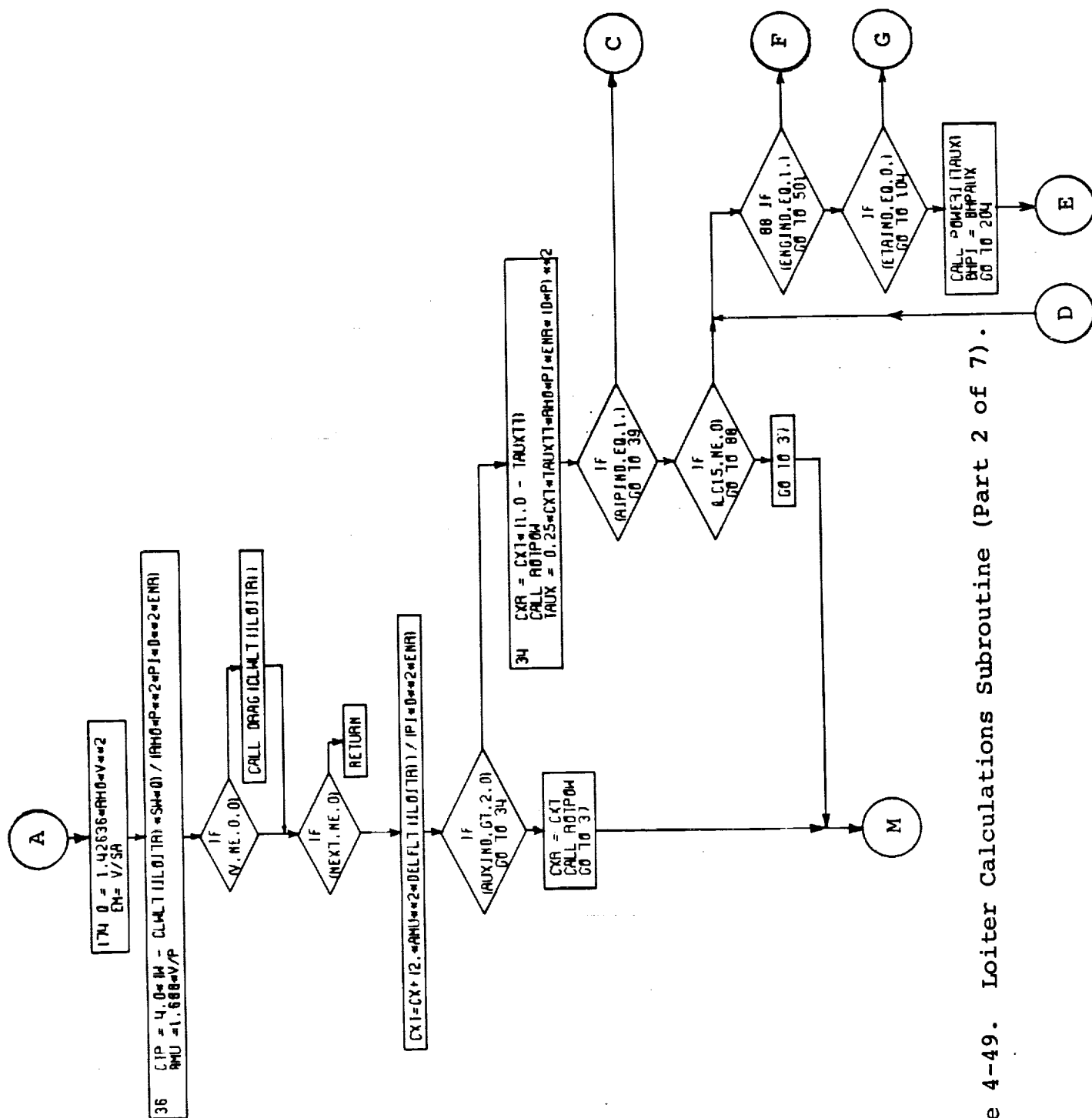
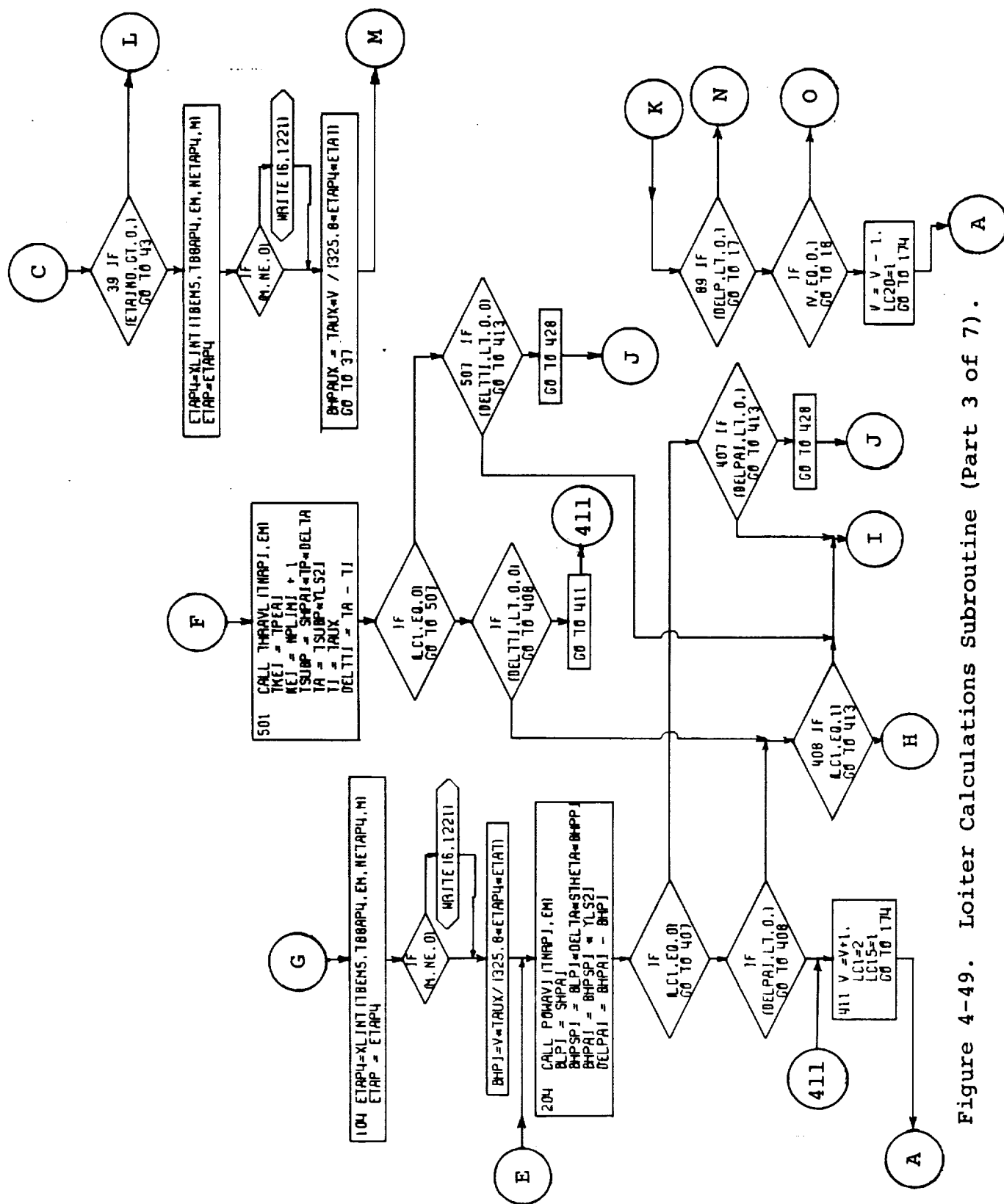


Figure 4-49. Loiter Calculations Subroutine (Part 2 of 7).



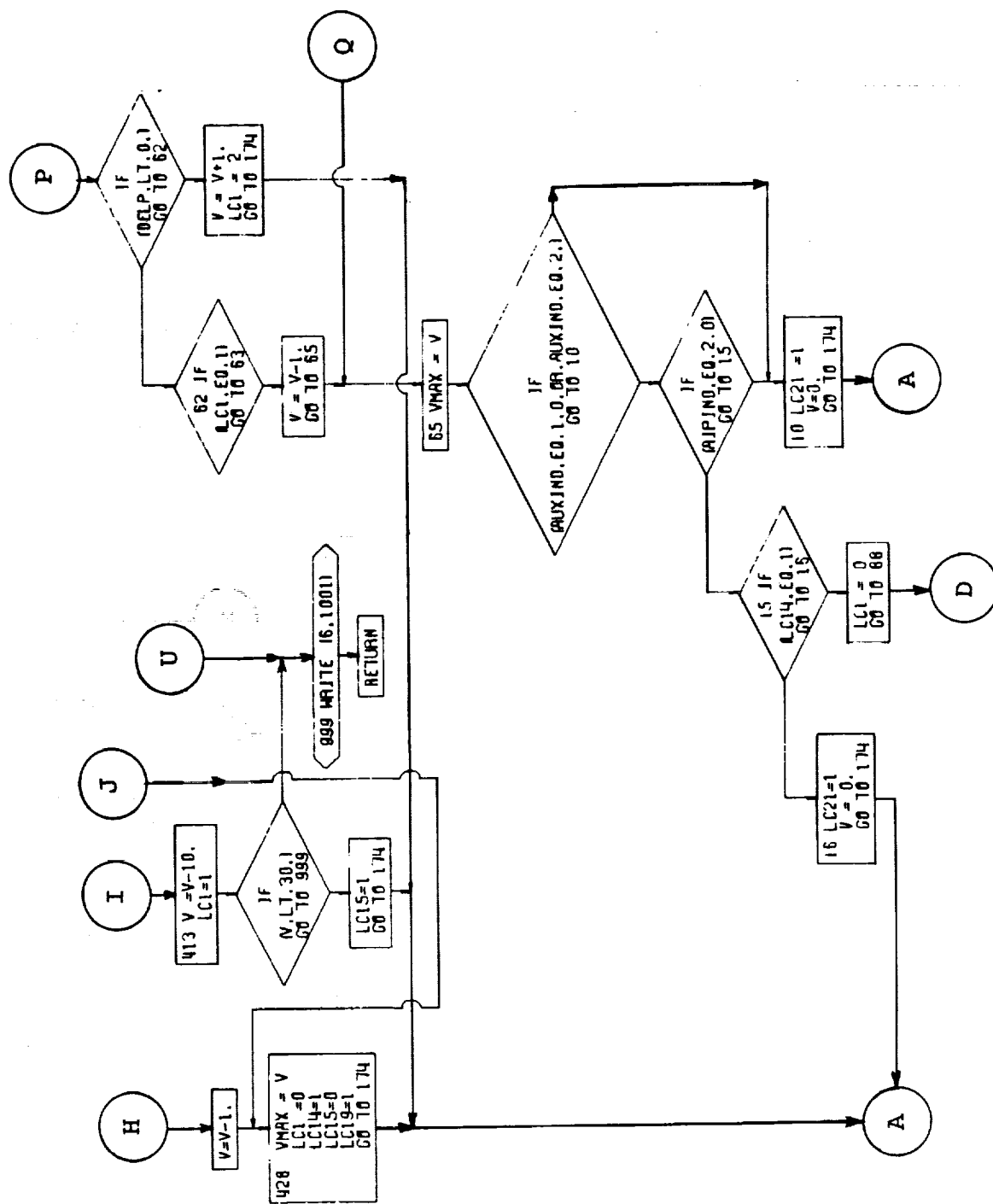


Figure 4-49. Loiter Calculations Subroutine (Part 4 of 7).

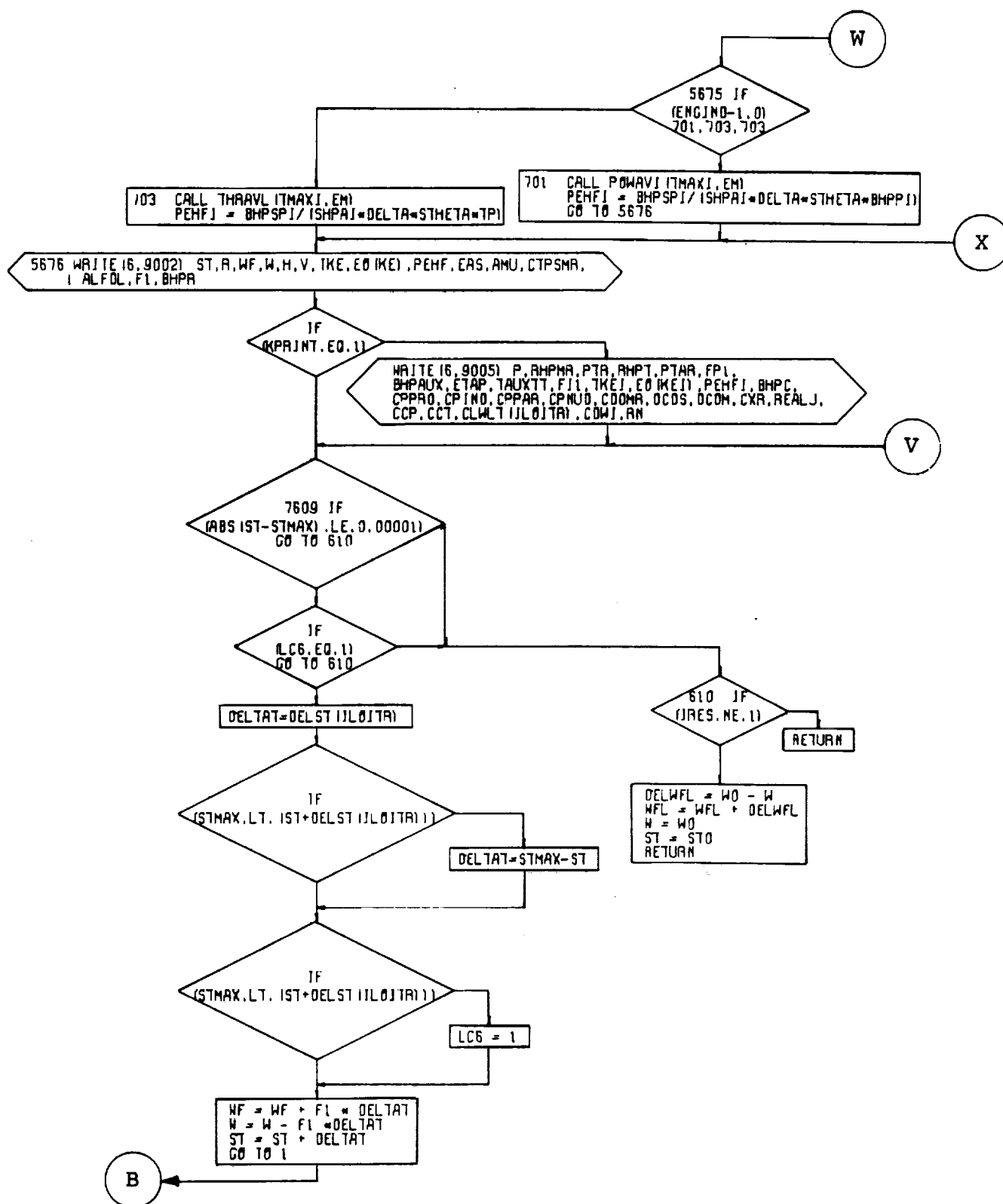


Figure 4-49. Loiter Calculations Subroutine (Part 7 of 7).

4.12.7 Change of Weight Subroutines

The seventh and eighth performance segments represent an incremental change in weight of fuel or payload. These options would be used to simulate refueling, unloading or loading of passengers, or a fuel drop. The input to the subroutines consists of the increment in weight and a corresponding increment in time. The fuel or payload weight which is added is not allowed to increase the aircraft weight to a value greater than the gross weight unless a performance case is being run and WGTIND = 1. Inputting a large value for the increment in weight will bring the aircraft weight up to gross weight if WGTIND = 0 or a sizing case is being run. Figures 4-50 and 4-51 are flow charts of these subroutines.

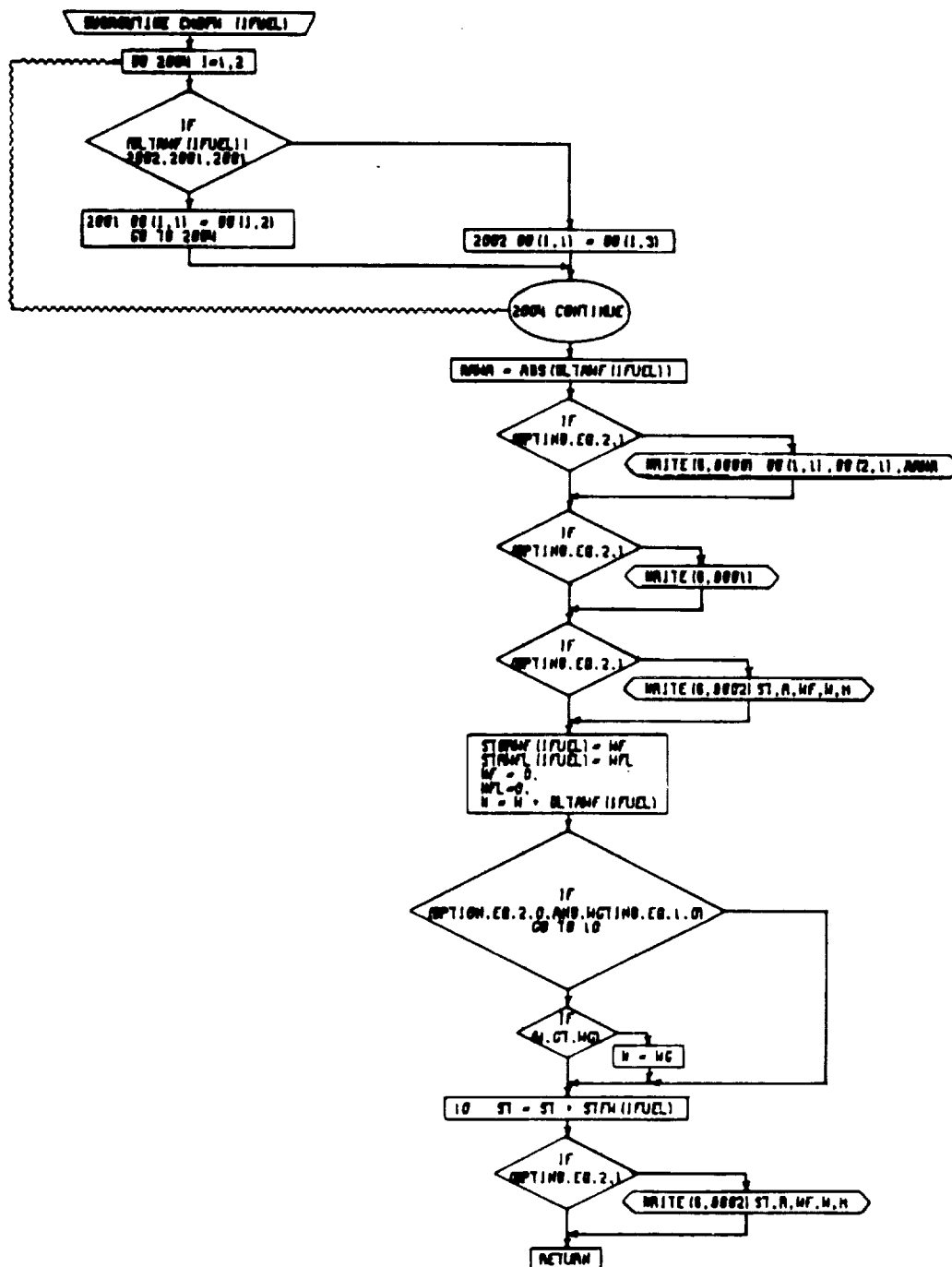


Figure 4-50. Change of Fuel Weight Subroutine, Flow Chart.



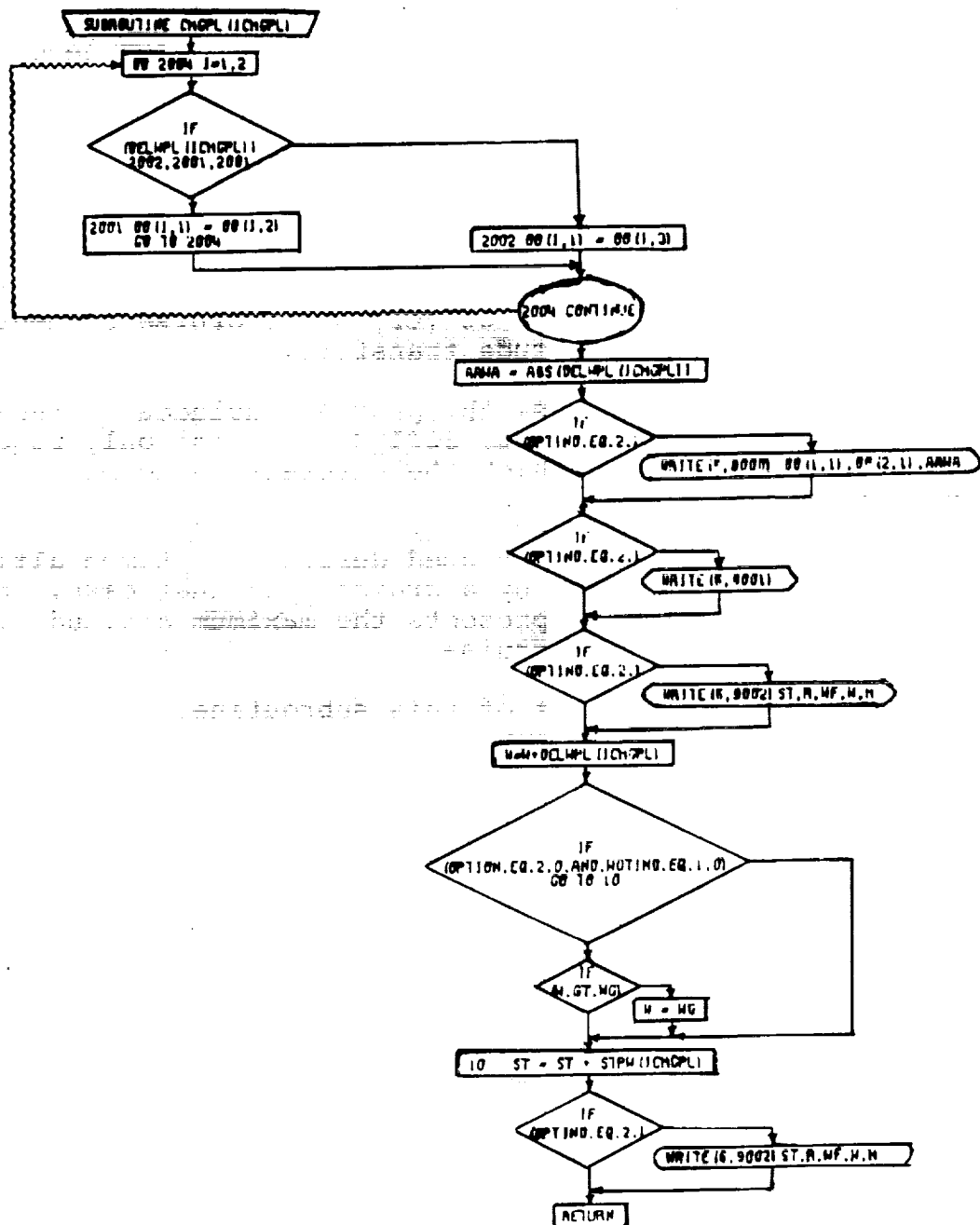


Figure 4-51. Change of Payload Weight Subroutine, Flow Chart.

4.12.8 Transfer Altitude

There are many different applications for which a discontinuous change in altitude may be desirable:

1. The flight profile may require takeoff at hot day, high altitude conditions followed by climb from sea level to specified altitude for standard day conditions.
2. It may be required that no credit be taken for range, fuel, or distance during descent (for example, Reference 9).
3. It may be required to study cruise speed at specified power at a series of different altitudes. This can be accomplished by a series of very short cruise segments interspersed with altitude transfers.

For these and other reasons, the program includes a transfer altitude segment, specified by SGTIND = 9. The only required input is the altitude to which the aircraft is to be transferred.

Transfer altitude may also be used during an optimum altitude search when it is followed by a cruise. In that case, the altitude which is input represents the maximum altitude permitted for the subsequent cruise.

Figure 4-52 is a flow chart of this subroutine.

5.0 PROGRAM INPUT

5.1 GENERAL

Input to the program is made by means of a standard set of input sheets. Although there are large quantities of possible input, necessitated by the requirement to keep the program flexible and general, the input sheets have been configured to give maximum visibility and reduce the tediousness of inputting the data. This has been accomplished through several means:

1. All input of a similar nature has been grouped together. Thus, all dimensional information is on the same input sheet, regardless of whether it is used in the size trends subroutine or elsewhere.
2. The input sheets have been color-coded to distinguish between the data required in the sizing option (OPTIND = 1) and the much smaller amount of data required for performance calculations (OPTIND = 2 or 3).
3. Footnotes on the input sheets call attention to input which is not required due to selection of one of the optional paths of computation.
4. For parametric studies where only one or two variables are being changed from case to case, a special supplementary input sheet may be used, thus reducing the quantity of paper work.

Altogether there are 31 different input sheets which can be loosely grouped into seven categories: general information, aircraft descriptive information, mission profile information, engine cycle information, rotor data, propeller data, and supplementary information. A specimen copy of each input sheet is included in this section. Descriptions of input variables and indicators are given in paragraphs 5.3.1 and 5.3.2. The use of the various input sheets is discussed below in paragraphs 5.1.1 and 5.1.2.

5.1.1 General Information

Input all primary program indicators (except those for specific mission segments, such as CRSIND), mission initial conditions, reserve fuel factors, and maneuver load factor.

5.1.2 Aircraft Description Information

5.1.2.1 Dimensional Information - Input characteristic geometric information for aircraft being studied.

5.1.2.2 Propulsion Information - Input data for numbers of primary and auxiliary independent propulsion engines, propellers, propeller cruise efficiencies, etc., and critical engine sizing conditions. There are three different input sheets for propulsion information: one for the primary engines (always used) and two for use with compound and auxiliary propulsion helicopter configurations. Of these two, one sheet is for propeller data and the other for auxiliary independent propulsion engine data.

5.1.2.3 Aerodynamics Information - Input aircraft drag characteristic and (in the case of compound and winged helicopters) wing section lift/drag characteristics.

5.1.2.4 Weight Information - Input the factors and constants for weight trends calculations.

5.1.3 Mission Profile Information

There are seven input sheets for mission profile information. They are:

1. Taxi Information
2. Takeoff, Hover, and Landing Information
3. Climb Information
4. Cruise Information
5. Descent Information
6. Loiter Information
7. Change of Weight and Transfer Altitude Information
(incorporating change of fuel weight, change of payload weight, and transfer altitude)

Each input variable on the mission profile sheets is represented by an array of ten input locations. The data for these locations is filled in sequentially by rows as the particular mission segment is used. For example, the first time that taxi is used in a particular case, the required input information is filled in on the first row of the input sheet. Data for the second taxi of a case is filled in on the second row, and so on. Thus, up to ten of any particular segment may be used in a case.

5.1.4 Engine Cycle Information

The engine cycle sheets may be used to input engine cycle data when one of the standard engine cycles is not used. The three engine cycle sheets are divided into standard performance information and nonstandard performance information. The standard performance data, of which there are two sheets, represent the performance of idealized engine cycles. These data are unlimited except for the effect of engine ratings, which are dictated by values of turbine temperature. The nonstandard

performance represents limiting values of fuel flow, torque, rpm and other nonstandard effects. It should be noted that auxiliary independent engine input data can be created from the HESCOMP engine cycle library data simply by the input of the applicable engine cycle IBM card deck, preceded and followed by a "66666" card. Nonstandard auxiliary independent engine performance is input using the sheet provided for that purpose.

5.1.5 Propeller Performance Data

The propeller performance sheets may be used to input data for a specific propeller when using $\eta_{pIND} = 1$. The data is input as a table of C_p as a function of C_T and J .

5.1.6 Rotor Performance Data

Rotor performance data can be input in two ways. If the short form "aero" performance computation method is used ($ROTIND = 1$), rotor data is input using the rotor "cycle" input sheet. Alternatively, rotor performance data may be input in "map" form ($ROTIND = 2, 3$) on the five rotor performance map input sheets. The first sheet is for hover performance data which is input as a table of C_{pH}/σ as a function of C_T/σ and M_{TIP} . The remaining four sheets are for cruise performance data which is input as a table of C_p/σ as a function of μ , C_T/σ , C_X/σ .

5.1.7 Supplementary Information

The supplementary input sheet may be used for the second and subsequent cases of a parametric study. For example, in the case of a tandem rotor helicopter, if the user wishes to change both the rotor overlap/diameter ratio (location 0132 - see dimensional information sheet) and the disc loading (location 0173 - see dimensional information sheet), these locations and their new values may be filled in on the supplementary input sheet.

Two typical problems, from input to output, are discussed in Section 7.6 of this manual.

GENERAL INFORMATION

TITLE CARD (72) (DIGITS)	7	10	13	16	19	22	25	28	31	34	37	40	42
	43	46	49	52	55	58	61	64	67	70	73	76	78

OPTION
INDICATORPRINT
INDICATORAERO-
DYNAMICS
INDICATORS1ST
ORDER
SIZE
TREND
AND
PRO-
PULSION
INDICATORS
(ALWAYS
INPUT)2ND
ORDER
SIZE
TREND
AND
PRO-
PULSION
INDICATORS
(OPTIONAL
INPUT)INITIAL
CONDITIONSFLIGHT PATH
CONTROL IN-
DICATORLIMITING
SPEEDMANEUVER
LOAD
FACTORRESERVE
FUEL
FACTORS

VARIABLE	UNIT	LOC.	VALUE
OPTIND		0001	

OPTIONAL PRINT		0002	
-------------------	--	------	--

DRGIND		0003	
OSWIND		0004	

CNFIND		0005	
AUXIND		0006	
RODMIND		0007	
FIXIND		0008	
ROTIND		0009	

SWIND	NOTE a	0010	
bWIND	NOTE a	0011	
AIPIND	NOTE b	0012	
ENGIND	NOTE c	0013	
FIXINDI	NOTE c	0014	
TRDIND	NOTE d	0015	
TRSIND	NOTE d	0016	
VTFIND	NOTE d	0017	
HTIND	NOTE d	0018	
MRPIND	NOTE d	0019	
FDIND	NOTE e	0020	
APHIND	NOTE e	0021	
ESCIND	NOTE f	0022	

WG ₀	LBS	0023	
h ₀	FT	0024	
R ₀	NM	0025	
t ₀	HR	0026	

hOPTIND		0027	
---------	--	------	--

M _{MO}		0028	
V _{MO}	KTS EAS	0029	
V _{DIVE}	KTS EAS	0030	

M _{LF}		0031	
-----------------	--	------	--

K _I		0032	
δW _F	LBS	0033	
K _{FF}		0034	

MISSION PROFILE INFORMATION

MAXIMUM OF 50 CONSECUTIVE SEGMENTS

VALUES OF SGTIND

0 = END OF MISSION

1 = TAXI

2 = T.O., HOVER, LAND

3 = CLIMB

4 = CRUISE

100 = END OF CASE

5 = DESCENT

6 = LOITER

7 = CHANGE FUEL

8 = CHANGE PAYLOAD

9 = TRANSFER ALTITUDE

	LOC.	VALUE		LOC.	VALUE
1st	0035		26th	0060	
2nd	0036		27th	0061	
3rd	0037		28th	0062	
4th	0038		29th	0063	
5th	0039		30th	0064	
6th	0040		31st	0065	
7th	0041		32nd	0066	
8th	0042		33rd	0067	
9th	0043		34th	0068	
10th	0044		35th	0069	
11th	0045		36th	0070	
12th	0046		37th	0071	
13th	0047		38th	0072	
14th	0048		39th	0073	
15th	0049		40th	0074	
16th	0050		41st	0075	
17th	0051		42nd	0076	
18th	0052		43rd	0077	
19th	0053		44th	0078	
20th	0054		45th	0079	
21st	0055		46th	0080	
22nd	0056		47th	0081	
23rd	0057		48th	0082	
24th	0058		49th	0083	
25th	0059		50th	0084	

NOTES: INPUT ONLY IF:

a. AUXIND = 2,4

b. AUXIND = 3,4

c. AIPIND = 2

d. CNFIND = 1

e. CNFIND = 2

f. FIXIND = 1

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY
THOSE ITEMS IN THE SHADED BLOCKS

HELICOPTER DIMENSIONAL INFORMATION

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY
THOSE ITEMS IN THE SHADED BLOCKS

NOTE	VARIABLE	UNIT	LOC.	VALUE
a	S_w	FT ²	0101	
b	w/s	PSF	0102	
c	b_w/D		0103	
d	AR		0104	
	$(t/c)_R$		0105	
	$(t/c)_T$		0106	
	$\Delta C/4$	DEG	0107	
	λ		0108	
e	C_F/C		0109	
	h'/h_F		0110	
f	C_{L_D}		0111	

WING

NOM = 1.0

	AR_{HT}		0112	
	l_{TH}'		0113	
	$(t/c)_{HT}$		0114	
g	$\bar{V}_H$		0115	
	λ_H		0116	
h	S_{HT}	FT ²	0117	

HOR.
TAIL

i	Y_{CL}	FT	0118	
i	ζ_2		0119	

AUX.
PROP.

(IF LOCATED ON WING)

	$\Delta S_{WET}/S_F$		0120	
	ΔS_{WET}	FT ²	0121	

GEN.

NOTE	VARIABLE	UNIT	LOC.	VALUE
	h_F	FT	0122	
	W_F	FT	0123	
	$(l/d)_p$		0124	
	$(l/d)_T$		0125	
	l_C	FT	0126	
	l_{RW}	FT	0127	
j	(X_M/l_B)		0128	
k	(l_{TB}/d_{TB})		0129	
k	(d_{TB}/d_{TB})		0130	
k	k_T STING		0131	
l	$(t_o/L)/D$		0132	
m	$(\Delta x_1/l_P)$		0133	
m	$(\Delta x_2/l_T)$		0134	

BODY

n	AR_{VT}		0135	
	λ_{VT}		0136	
	$(t/c)_{VT}$		0137	
p	ζ_{VT}		0138	
	K_Z		0139	
q	C_{LDES}		0140	
q	V_{DES}	KTS TAS	0141	

VERT.
TAIL

	γ_1		0142	
	γ_2		0143	
	γ_3		0144	
	(l_{AIP}/l_e)		0145	

PRIM.
ENG.
NAC.AUX.
IND.
ENG.
NAC.

r	γ_4		0146	
r	γ_5		0147	
r	γ_6		0148	
r	(l_{AIA}/l_{eA})		0149	
r	$\Delta S/S_{STR}$		0150	
r	b_{NS}/d_{NI}		0151	

NOTES: INPUT NOT NECESSARY WHEN:

- | | |
|--------------------|-----------------|
| a. S_w IND = 2,3 | j. MRPIND = 1,2 |
| b. S_w IND = 1,3 | k. CNFIND = 2 |
| c. b_w IND = 2,3 | l. FDMIND = 3 |
| d. b_w IND = 1,3 | m. FDMIND = 2 |
| e. AUXIND = 1,3 | n. VTFIND = 2 |
| f. S_w IND = 1,2 | p. VTFIND = 3 |
| g. HTIND = 1 | q. VTFIND = 1 |
| h. HTIND = 2 | r. AIPIND = 2 |
| i. b_w IND = 1,2 | |

HELICOPTER DIMENSIONAL INFORMATION

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY
THOSE ITEMS IN THE SHADED BLOCKS

FWD.
ROTOR
PYLON
†

NOTE	VARIABLE	UNIT	LOC.	VALUE
	$(t/c)R_F$		0152	
	$(t/c)T_F$		0153	
	AR_{FP}		0154	
	λ_{FP}		0155	
	h_{p1}	FT	0156	

AFT
ROTOR
PYLON

	$(t/c)R_A$		0157	
	$(t/c)T_A$		0158	
	AR_{AP}		0159	
	λ_{AP}		0160	
a	h_{p2}	FT	0161	
b	g/s		0162	

† ~ IN THE CASE OF THE SINGLE ROTOR HELICOPTER,
THIS BECOMES THE MAIN ROTOR PYLON

NOTES: INPUT NOT NECESSARY IF:

- a. APHIND = 2
- b. APHIND = 1

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ROTOR DIMENSIONAL DATA FOR SIZING MAIN ROTOR(S)

ROTOR MAP NO.	0170
ROTOR CYCLE NO.	0171

NOTE	VARIABLE	UNIT	LOC.	VALUE
	N_R		0172	
a	W/A	PSF	0173	
b	D_{MR}	FT	0174	
c	σ_{MR}		0175	
	b_{MR}		0176	
	θ_{TMR}	DEG	0177	
	$X_{C_{MR}}$		0178	
	X_{MR}		0179	
	$(t/c)_{.25R}$		0180	
	V_{TIP}	FPS	0181	

ROTOR
CHARACTERISTICS

HOVER
COND.
FOR
ROTOR
SIZING

d	$(C_T/\sigma)_H$		0182	
d	T/W		0183	

CRUISE
COND
FOR
ROTOR
SIZING

	$V_{KT(c)}$	KTS TAS	0184	
	$h_C(c)$	FT	0185	
	$\Delta TINC$	$^{\circ}F$	0186	
d	$(C_T/\sigma)_{CR}$		0187	
	g REQMT		0188	
	g (ROTOR)		0189	
	(ROTOR N LOADING)		0190	

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY THOSE
ITEMS IN THE SHADED BLOCKS.

ROTOR
VERTICAL R/C
EFFICIENCY
FACTORS

	V_{CEH1}	0191
	V_{CEH2}	0192

† IF NO. OF ROTORS = 1, MUST ADD
TAIL ROTOR SIZING SHT.

ROTOR
CLIMB &
DESCENT
EFFICIENCIES

	K_P CLIMB	0193
	K_P DESCENT	0194

NOTES: INPUT NOT NECESSARY WHEN

a.	RDIND = 1,3
b.	RDIND = 2,4
c.	RDIND = 3,4
d.	RDIND = 1,2

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COMPUTER PROGRAM B-91

ROTOR DIMENSIONAL DATA FOR SIZING TAIL ROTOR (CNFIND = 1)

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY THOSE ITEMS IN THE SHADED BLOCKS.

NOTE	VARIABLE	UNIT	LOC	VALUE
a	(T/A) NET	PSF	0200	
b	D _{TR}	FT	0201	
c	σ _{TR}		0202	
	b _{TR}		0203	
	θ _{T_{TR}}	DEG	0204	
	x _{C_{TR}}		0205	
	x _{TR}		0206	
	V _{TTR}	FPS	0207	

TAIL
ROTOR
CHARACTERISTICS

NOTES: a. INPUT NOT NECESSARY WHEN TRDIND = 1.2
b. INPUT NOT NECESSARY WHEN TRDIND = 1.3
c. INPUT NOT NECESSARY WHEN TRSIND = 2
d. INPUT NOT NECESSARY WHEN TRSIND = 1
e. IF OPTIND = 1, (C_T / C_{T_G})_{NET} MAY BE INPUT AS 1.00, AND A VALUE OF C̄ INPUT, PROGRAM WILL THEN CALCULATE A VALUE OF (C_T / C_{T_G})_{NET}. IF VALUE OF (C_T / C_{T_G})_{NET} > 1.0 INPUT, IT IS NOT NECESSARY TO INPUT C̄.

TAIL
ROTOR
SIZING
COND.

NOM. = 1.0

d	(C _T / σ) _{DES (H)}		0208	
	YAW ACCEL (ψ̈)	RAD 2 SEC	0209	
	YAW RATE (ψ̇)	RAD SEC	0210	
e	(C _T / C _{T_G}) _{NET}		0211	
e	C		0212	
d	K _{ZZZ}		0213	
	g _{MPR/TR}	FT	0214	
d	K _{TRS}		0215	

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HELICOPTER PROPULSION INFORMATION REQUIRED FOR PRIMARY ENGINE SIZING

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY THOSE ITEMS IN THE SHADED BLOCKS

PRIMARY ENGINE	CYCLE NO.
	0217

NOTE	VARIABLE	UNIT	LOC	VALUE
a	SH _P [*]	HP	0218	

N _P	0219
----------------	------

PRIMARY
ENGINE
DATA

	XMSNIND		0220
	SHP _{MRX} /SHP* MR		0221
b	KALT PAYL		0222
	η_T		0223
	Δ SHP ACC	HP	0224

MAIN
XMSN
&
ACC
DATA

SH _P TR _X /TR _P [*]	0225
SH _P AR _X /SH _P [*] AUX	0226

T. ROTOR
&
AUX DRIVE SYS
XMSN DATA

- NOTES: a. INPUT NOT NECESSARY WHEN FIXIND = 1
b. INPUT NOT NECESSARY WHEN XMSNIND = 1,2
c. INPUT NOT NECESSARY WHEN AUXIND = 1,2
d. INPUT NOT NECESSARY WHEN AUXIND = 1,3

NOTE	VARIABLE	UNIT	LOC	VALUE
	h _{TO(H)}	FT	0227	
	(T/A) ₀		0228	
	ΔT IN TO (H)	°F	0229	
	($\frac{N_{II}}{N_{II\text{MAX}}}$) TO		0230	
	N _{PSD}		0231	
	SH _P _E /SH _P [*]		0232	
	(V _{R/C}) ³ D	FPM	0233	

T/O
COND
FOR
ENG
SIZING

	POWIND		0234	
	h _C	FT	0235	
	V _C	KTS TAS	0236	
	ΔT INCE	°F	0237	
	(N _{II} /N _{II\text{MAX}}) C		0238	
c	(T _{AUX} /T _{TOT C})		0239	
d	C _{LDP}		0240	
	(N _{PSD}) ³ C		0241	

CRUISE
COND
FOR
ENG
SIZING

HESCOMP HELICOPTER SIZING AND PERFORMANCE
COMPUTER PROGRAM B-91

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HELICOPTER PROPULSION INFORMATION REQUIRED FOR SEPARATE AUXILIARY CRUISE ENGINE SIZING (AIPIND = 2)

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY THOSE
ITEMS IN THE SHADED BLOCKS.

AUX PROPULSION ENGINE CYCLE NO.		0242
------------------------------------	--	------

NOTE	VARIABLE	UNIT	LOC	VALUE
a	SHp^*_i	HP	0243	
a	F_N^*	LBS	0244	
	N_P		0245	

AUX
PROPULSION
ENGINE
DATA

	POWIND		0246	
	$(N_{II}/N_{II_{MAX}})_i$		0247	

CRUISE
CONDITIONS
FOR AUX.
PROPULSION
ENGINE
SIZING

NOTES: a. INPUT NOT NECESSARY WHEN FIXINDI = 1.

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PROPELLER DATA REQUIRED FOR COMPOUND HELICOPTER AUXILIARY PROPULSION (T/SHAFT ENGINES)

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY THOSE ITEMS IN THE SHADED BLOCKS.

η_{p4} : CRUISE & LOITER

NO. OF PAIRS IN η_{p4} TABLE	0261
-----------------------------------	------

VALUES OF EFFICIENCY

NOTE	VARIABLE	UNIT	LOC	VALUE
CLIMB	η_{p3}		0264	
DESCENT	η_{p5}		0265	

NOTE	VARIABLE	UNIT	LOC	VALUE
	NO. OF PROPS		0248	
	V_{TAR}	FPS	0249	
	DIA	FT	0250	
	XAR		0251	
	$\eta_{T AUX}$		0252	

η_p IND = 0

NOTE	VARIABLE	UNIT	LOC	VALUE
	AF/BLADE		0257	
	NO. OF BLADES		0258	

η_p IND	0253
--------------	------

0 = INPUT η_p 'S
1 = INPUT PROP TABLE
2 = PROG CALC PROP PERF.

η_{p5}	0266
PROP TABLE NO.	0260

NOTE: ALSO INPUT PROP TABLE (LOCS. 1700 → 2142)

NO. OF BLADES	0268
C_{L-1}	0269
η_{p6}	0265

VALUES OF η_{p4}			
LOC	VALUE	LOC	VALUE
0262		0272	
0263		0273	
0264		0274	
0265		0275	
0266		0276	
0267		0277	
0268		0278	
0269		0279	
0270		0280	
0271		0281	

VALUES OF η_{p4}

HELICOPTER AERODYNAMICS INFORMATION

NOTE: WHEN OPTIND = 2 CONSIDER ONLY THOSE ITEMS IN THE SHADED BLOCKS

NOTE	VARIABLE	LOC.	VALUE
a	C_{DVT}	0301	
a	C_{DHT}	0302	
b	C_{DAP}	0303	
	C_{DFP}	0304	
	C_{DCSMR}	0305	
	C_{DSHMR}	0306	
a	C_{DCSTR}	0307	
a	C_{DSHTR}	0308	
	C_{DN}	0309	
	C_{DNI}	0310	
	C_{DNS}	0311	
c	(G_w/F_e)	0312	
c	K_{FED}	0313	
d	e	0314	
a	TFEF	0315	
	$\Delta F_e FT^2$	0316	

NOTE	VARIABLE	LOC.	VALUE
a	K_{VT}	0317	
a	K_{HT}	0318	
b	K_{AP}	0319	
	K_{FP}	0320	
	K_{HPIM}	0321	
a	K_{HPIT}	0322	
	K_N	0323	
	K_{NI}	0324	
	K_{NS}	0325	
	K_F	0326	
	K_W	0327	

$(Re/l)_i$	0328	
------------	------	--

$C_{l_\alpha} \text{ RAD}^{-1}$	0329	
---------------------------------	------	--

WING PROFILE DRAG AS FUNCTION OF C_L

NO. OF PAIRS IN TABLE	0330	
--------------------------	------	--

NOTE: C_{LW} , C_{DWi} INPUT
NOT NECESSARY
WHEN AUXIND = 1,3

	$C_{LW(1)}$	0331	
	$C_{LW(2)}$	0332	
	$C_{LW(3)}$	0333	
	$C_{LW(4)}$	0334	
	$C_{LW(5)}$	0335	
	$C_{LW(6)}$	0336	
	$C_{LW(7)}$	0337	
	$C_{LW(8)}$	0338	

	$C_{DWi(1)}$	0339	
	$C_{DWi(2)}$	0340	
	$C_{DWi(3)}$	0341	
	$C_{DWi(4)}$	0342	
	$C_{DWi(5)}$	0343	
	$C_{DWi(6)}$	0344	
	$C_{DWi(7)}$	0345	
	$C_{DWi(8)}$	0346	

NOTES: a. INPUT NOT NECESSARY WHEN CNFIND = 2
b. INPUT NOT NECESSARY WHEN CNFIND = 1
c. INPUT NOT NECESSARY WHEN DRGIND = 1
d. INPUT NOT NECESSARY WHEN OSWIND = 1

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HELICOPTER WEIGHT INFORMATION

NOTE: WHEN OPTIND = 2 OR 3 CONSIDER ONLY THOSE ITEMS IN THE SHADED BLOCKS

VARIABLE	LOC.	VALUE
* OWE	2601	
W _{FE}	2602	
W _{FUL}	2603	
** W _{PL}	2604	

INCREMENTAL GROUP WTS. NOM = 0

VARIABLE	LOC.	VALUE
ΔW_{FC}	2605	
ΔW_P	2606	
ΔW_{ST}	2607	

VARIABLE	LOC.	VALUE
RM _i	2608	
W _i	2609	
W _o	2610	
d _i	2611	
d _o	2612	

GROUP WEIGHT INFORMATION

FLIGHT CONTROLS

k _{CC}	2613	
k _{RC}	2614	
k _{SC}	2615	
k _{FW}	2616	
k _{TM}	2617	
k _{SAS}	2618	
k _{RCA}	2619	
k _{SCA}	2620	
k _{MC}	2621	

* OWE IS NOT NECESSARY WHEN
OPTIND = 1, 2

** W_{PL} IS NOT NECESSARY
WHEN OPTIND = 2

STRUCTURAL

k _B	2622	
$\Delta C.G.$	2623	
k _{LG}	2624	
k _{MG}	2625	
k _{WW}	2626	
LF	2627	
k _{WS}	2628	
k _{WP}	2629	
k _{HT}	2630	
k _{CLF}	2631	
k _{NAC}	2632	
k _{AIP}	2633	
k _{NACA}	2634	
k _{AIA}	2635	
k _{NS}	2636	

PROPULSION

k _{PRB}	2637	
k _{RBF}	2638	
k _{PH}	2639	
k _{amd}	2640	
k _{BLFD}	2641	
k _{TR}	2642	
k _{AR}	2643	
k _{PA}	2644	
k _{VTAR}	2645	
k _{PDS}	2646	
k _{PDSZ}	2647	
k _{TRDS}	2648	
k _{ADS}	2649	
k _{ADSZ}	2650	
k _{FS}	2651	
k _{PEI}	2652	
k _{AEI}	2653	

MULTIPLICATIVE FACTORS
NOMINALLY = 1.0

K ₁	2654	
K ₂	2655	
K ₃	2656	
K ₄	2657	
K ₅	2658	

K ₆	2659	
K ₇	2660	
K ₈	2661	
K ₉	2662	
K ₁₀	2663	
K ₁₁	2664	

K ₁₂	2665	
K ₁₃	2666	
K ₁₄	2667	
K ₁₅	2668	
K ₁₆	2669	
K ₁₇	2670	
K ₁₈	2671	
K ₁₉	2672	
K ₂₀	2673	

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WEIGHT-BALANCE INFORMATION (REQUIRED ONLY WHEN MRPIND > 0)

VARIABLE	LOC.	VALUE
(X_{CGF}/l_B)	2678	
ΔCG_R	2679	
(X_{NG}/l_B)	2680	
(X_{MG}/l_B)	2681	
(X_{PE}/l_B)	2682	
$(X_{PDS}/\Delta X_{PDS})$	2683	
(X_{AV}/l_B)	2684	

VARIABLE	LOC.	VALUE
(X_{FURN}/l_B)	2685	
(X_{APU}/l_B)	2686	
(X_{AE}/l_B)	2687	
$(X_{ADS}/\Delta X_{ADS})$	2688	
(X_{AR}/l_{TB})	2689	
(X_{SC}/l_B)	2690	
(X_{ASC}/l_B)	2691	

VARIABLE	LOC.	VALUE
W_{AV}	2692	
W_{FURN}	2693	
W_{APU}	2694	
K_{FULS}	2695	
K_{TBBS}	2696	

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**HESCOMP HELICOPTER SIZING AND PERFORMANCE
COMPUTER PROGRAM B-91**

BOEING VERTOL COMPANY
A DIVISION OF THE BOEING COMPANY

ROTOR LIMITS INFORMATION

NUMBER OF C_X/σ	LOC	VALUE	VALUES OF C_X/σ		VALUES OF μ	
			$(C_X/\sigma)_1$	0349	μ_1	0354
			$(C_X/\sigma)_2$	0350	μ_2	0355
			$(C_X/\sigma)_3$	0351	μ_3	0356
			$(C_X/\sigma)_4$	0352	μ_4	0357
			$(C_X/\sigma)_5$	0353	μ_5	0358
					μ_6	0359
					μ_7	0360

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VALUES OF C_T/σ

	$(C_X/\sigma)_1 =$		$(C_X/\sigma)_2 =$		$(C_X/\sigma)_3 =$		$(C_X/\sigma)_4 =$		$(C_X/\sigma)_5 =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$\mu_1 =$	0361		0368		0375		0382		0389	
$\mu_2 =$	0362		0369		0376		0383		0390	
$\mu_3 =$	0363		0370		0377		0384		0391	
$\mu_4 =$	0364		0371		0378		0385		0392	
$\mu_5 =$	0365		0372		0379		0386		0393	
$\mu_6 =$	0366		0373		0380		0387		0394	
$\mu_7 =$	0367		0374		0381		0388		0395	

HESCOMP HELICOPTER SIZING AND PERFORMANCE
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BOEING VERTOL COMPANY
A DIVISION OF THE BOEING COMPANY

ATMOSPHERE TEMPERATURE

NO. OF PAIRS	1650
--------------	------

NOTE: THIS TABLE IS NOT
NECESSARY IF ATMIND
IS NEVER SET TO 2

h_1	1651	
h_2	1652	
h_3	1653	
h_4	1654	
h_5	1655	
h_6	1656	
h_7	1657	
h_8	1658	
h_9	1659	
h_{10}	1660	

θ_1	1661	
θ_2	1662	
θ_3	1663	
θ_4	1664	
θ_5	1665	
θ_6	1666	
θ_7	1667	
θ_8	1668	
θ_9	1669	
θ_{10}	1670	

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TAXI INFORMATION

SGTIND = 1

$\Delta T_{IN} (^{\circ}F)$
(NOTE a)

$(N_{II}/N_{II MAX})$
(PRIM ENG)

ATMIND	
LOC	VALUE
1 st	0401
2 nd	0402
3 rd	0403
4 th	0404
5 th	0405
6 th	0406
7 th	0407
8 th	0408
9 th	0409
10 th	0410

LOC	VALUE
0421	
0422	
0423	
0424	
0425	
0426	
0427	
0428	
0429	
0430	

LOC	VALUE
0441	
0442	
0443	
0444	
0445	
0446	
0447	
0448	
0449	
0450	

0 = STD ATMOSPHERE
1 = STD + ΔT_{IN}
2 = ARBITRARY θ (h)

t_T (HR)	
LOC	VALUE
1 st	0411
2 nd	0412
3 rd	0413
4 th	0414
5 th	0415
6 th	0416
7 th	0417
8 th	0418
9 th	0419
10 th	0420

K_{FI}	
LOC	VALUE
0431	
0432	
0433	
0434	
0435	
0436	
0437	
0438	
0439	
0440	

NOTE: a. INPUT NOT NECESSARY WHEN ATMIND = 0, 2

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SHEET NO.	CASE NO.
OF	

TAKEOFF, HOVER, AND LANDING INFORMATION (SGTIND = 2)

$(N_{II}/N_{II,MAX})$
(PRIMARY ENG)

TOLIND

LOC	VALUE
0461	
0462	
0463	
0464	
0465	
0466	
0467	
0468	
0469	
0470	

1st
2nd
3rd
4th
5th
6th
7th
8th
9th
10th

1 = INPUT T/W, V_{R/C}
2 = INPUT PEHF, V_{R/C}
3 = INPUT T/W, V_{R/C}, h_{BF/D}
4 = INPUT PEHF, V_{R/C}, h_{BF/D}

h_{BF/D} (NOTE c)

LOC	VALUE
0471	
0472	
0473	
0474	
0475	
0476	
0477	
0478	
0479	
0480	

1st
2nd
3rd
4th
5th
6th
7th
8th
9th
10th

ATMIND

LOC	VALUE
0481	
0482	
0483	
0484	
0485	
0486	
0487	
0488	
0489	
0490	

0 = STANDARD ATMOSPHERE
1 = STD + ΔT IN
2 = ARBITRARY θ (h)

PEHF
(NOTE d)

LOC	VALUE
0491	
0492	
0493	
0494	
0495	
0496	
0497	
0498	
0499	
0500	

ΔT IN (°F)
(NOTE a)

LOC	VALUE
0501	
0502	
0503	
0504	
0505	
0506	
0507	
0508	
0509	
0510	

T/W
(NOTE b)

LOC	VALUE
0521	
0522	
0523	
0524	
0525	
0526	
0527	
0528	
0529	
0530	

V_{R/C} (FPM)

LOC	VALUE
0511	
0512	
0513	
0514	
0515	
0516	
0517	
0518	
0519	
0520	

Δ_T (HR)

LOC	VALUE
0531	
0532	
0533	
0534	
0535	
0536	
0537	
0538	
0539	
0540	

t_H (HR)

LOC	VALUE
0551	
0552	
0553	
0554	
0555	
0556	
0557	
0558	
0559	
0560	

NOTES: a. INPUT NOT NECESSARY WHEN ATMIND = 0, 2
b. INPUT NOT NECESSARY WHEN TOLIND = 2, 4
c. INPUT NOT NECESSARY WHEN TOLIND = 3, 4
d. INPUT NOT NECESSARY WHEN TOLIND = 1, 3

HESCOMP HELICOPTER SIZING AND PERFORMANCE
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SHEET NO.	CASE NO.
OF	

CLIMB INFORMATION (SGTIND = 3)

T_{AUX}/T_{TOT}
(NOTE b)

$N_{II}/N_{II} \text{ MAX}$
(AUX ENG)

$N_{II}/N_{II} \text{ MAX}$
(PRIMARY ENG)

POWIND

$\Delta T_{in} (^{\circ}F)$
(NOTE a)

ATMIND

CLMIND

LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
0671		0681		0691		0611		0631		0651		0671	
0672		0682		0692		0612		0632		0652		0672	
0673		0683		0693		0613		0633		0653		0673	
0674		0684		0694		0614		0634		0654		0674	
0675		0685		0695		0615		0635		0655		0675	
0676		0686		0696		0616		0636		0656		0676	
0677		0687		0697		0617		0637		0657		0677	
0678		0688		0698		0618		0638		0658		0678	
0679		0689		0699		0619		0639		0659		0679	
0680		0690		0600		0620		0640		0660		0680	

1 = MAX R/C
2 = CONSTANT EAS
3 = CONSTANT MACH
4 = CONSTANT TAS

0 = STANDARD ATMOSPHERE
1 = STD + ΔT_{in}
2 = ARBITRARY θ (h)

0 = MAX POWER
1 = MIL POWER
2 = NORMAL POWER

C_{LWING}
(NOTE d)

Δh (FT)

h_{MAX} (FT)

ΔF_{eCL} (FT²)

$N_{PSD_{CL}}$

$N_{PSD_{iCL}}$

LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
0681		0691		0601		0621		0641		0661		0681	
0682		0692		0602		0622		0642		0662		0682	
0683		0693		0603		0623		0643		0663		0683	
0684		0694		0604		0624		0644		0664		0684	
0685		0695		0605		0625		0645		0665		0685	
0686		0696		0606		0626		0646		0666		0686	
0687		0697		0607		0627		0647		0667		0687	
0688		0698		0608		0628		0648		0668		0688	
0689		0699		0609		0629		0649		0669		0689	
0690		0600		0610		0630		0650		0670		0690	

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NOTES: a. INPUT NOT NECESSARY WHEN ATMIND = 0, 2 c. INPUT NOT NECESSARY WHEN CLMIND = 1
b. INPUT NOT NECESSARY WHEN AUXIND = 1, 2 d. INPUT NOT NECESSARY WHEN AUXIND = 1, 3

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SHEET NO. OF	CASE NO.
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CRUISE INFORMATION (SGTIND = 4)

	CRSIND		ATMIND		$\Delta T_{IN} (^{\circ}F)$ (NOTE a)		POWIND (NOTE b)		$N_{II}/N_{II} \text{ MAX}$ (PRIMARY ENG)		$N_{II}/N_{II} \text{ MAX}$ (AUX ENG)		T_{AUX}/T_{TOT} (NOTE c)	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
1 st	0721		0741		0761		0781		0801		0821		0841	
2 nd	0722		0742		0762		0782		0802		0822		0842	
3 rd	0723		0743		0763		0783		0803		0823		0843	
4 th	0724		0744		0764		0784		0804		0824		0844	
5 th	0725		0745		0765		0785		0805		0825		0845	
6 th	0726		0746		0766		0786		0806		0826		0846	
7 th	0727		0747		0767		0787		0807		0827		0847	
8 th	0728		0748		0768		0788		0808		0828		0848	
9 th	0729		0749		0769		0789		0809		0829		0849	
10 th	0730		0750		0770		0790		0810		0830		0850	

1 = SPECIFIED POWER
2 = CONSTANT TAS
3 = BEST NMPP
4 = 99% BEST NMPP
5 = BEST NMPP,
CONST. W/ δ
6 = 99% BEST NMPP,
CONST W/ δ

0 = MAX POWER
1 = MIL POWER
2 = NORMAL POWER

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	V_{IN} OR HEADWIND (NOTE d)		C_{LWING} (NOTE e)		$\Delta R(NM)$		$R_{MAX}(NM)$		$\Delta F_{eCR}(FT^2)$		$N_{PSD CR}$		$N_{PSD iCR}$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
1 st	0731		0751		0771		0791		0811		0831		0851	
2 nd	0732		0752		0772		0792		0812		0832		0852	
3 rd	0733		0753		0773		0793		0813		0833		0853	
4 th	0734		0754		0774		0794		0814		0834		0854	
5 th	0735		0755		0775		0795		0815		0835		0855	
6 th	0736		0756		0776		0796		0816		0836		0856	
7 th	0737		0757		0777		0797		0817		0837		0857	
8 th	0738		0758		0778		0798		0818		0838		0858	
9 th	0739		0759		0779		0799		0819		0839		0859	
10 th	0740		0760		0780		0800		0820		0840		0860	

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NOTES: a. INPUT NOT NECESSARY WHEN ATMIND = 0, 2
b. INPUT NOT NECESSARY WHEN CRSIND = 3
c. INPUT NOT NECESSARY WHEN AUXIND = 1, 3
d. INPUT V_{IN} WHEN CRSIND = 2
e. INPUT NOT NECESSARY WHEN CRSIND = 3 THRU 6

HESCOMP HELICOPTER SIZING AND PERFORMANCE

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BOEING VERTOL COMPANY

A DIVISION OF THE BOEING COMPANY

DESCENT INFORMATION (SGTIND = 5)

	DESIND	RMAXIND	ATMIND	ΔT IN (°F) (NOTE a)	R/D (FPM)	N _{II} /N _{II} MAX (PRIMARY ENG)	N _{II} /N _{II} MAX (AUX ENG)	T _{AUX} /T _{TOT} (NOTE b)
1 ST	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE
2 ND	0871	0891	0811	0831	0851	0871	0891	1011
3 RD	0872	0892	0812	0832	0852	0872	0892	1012
4 TH	0873	0893	0813	0833	0853	0873	0893	1013
5 TH	0874	0894	0814	0834	0854	0874	0894	1014
6 TH	0875	0895	0815	0835	0855	0875	0895	1015
7 TH	0876	0896	0816	0836	0856	0876	0896	1016
8 TH	0877	0897	0817	0837	0857	0877	0897	1017
9 TH	0878	0898	0818	0838	0858	0878	0898	1018
10 TH	0879	0899	0819	0839	0859	0879	0899	1019
	0880	0900	0820	0840	0860	0880	1000	1020

1 = CONSTANT TAS

2 = CONSTANT EAS

3 = CONSTANT MACH NO.

0 = TERMINAL RANGE SPECIFIED

1 = TERM. RANGE CHK'D, BUT NOT MATCHED

2 = DESCENT TO SPECIFIED MIN ALT.

3 = SPIRAL DESCENT PATH

0 = STD ATMOSPHERE

1 = STD + ΔT_{IN}

2 = ARBITRARY θ(h)

	MACH OR EAS OR TAS	C _{LW} (NOTE c)	h _{MIN} (FT)	R _{MAX} (NM)	ΔF _{eDSC} (FT ²)	N _{PSD DSC}	N _{PSD I_{DSC}}
1 ST	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE	LOC VALUE
2 ND	0881	0901	0841	0861	0881	1001	1021
3 RD	0882	0902	0842	0862	0882	1002	1022
4 TH	0883	0903	0843	0863	0883	1003	1023
5 TH	0884	0904	0844	0864	0884	1004	1024
6 TH	0885	0905	0845	0865	0885	1005	1025
7 TH	0886	0906	0846	0866	0886	1006	1026
8 TH	0887	0907	0847	0867	0887	1007	1027
9 TH	0888	0908	0848	0868	0888	1008	1028
10 TH	0889	0909	0849	0869	0889	1009	1029
	0890	0910	0850	0870	0890	1010	1030

NOTES: a. INPUT NOT NECESSARY WHEN ATMIND = 0, 2

b. INPUT NOT NECESSARY WHEN AUXIND = 1, 2

c. INPUT NOT NECESSARY WHEN AUXIND = 1, 3

HESCOMP HELICOPTER SIZING AND PERFORMANCE
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SHEET NO.	CASE NO.
OF	

LOITER INFORMATION (SGTIND = 6)

ATMIND		ΔT_{IN} (°F) (NOTE a)		$(N_{II}/N_{II MAX})$ (PRIMARY ENG)		$(N_{II}/N_{II MAX})$ (AUX ENG)		T_{AUX}/T_{TOT} (NOTE b)	
LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
1 st	1031	1051		1071		1091		1111	
2 nd	1032	1062		1072		1092		1112	
3 rd	1033	1053		1073		1093		1113	
4 th	1034	1054		1074		1094		1114	
5 th	1036	1056		1075		1095		1115	
6 th	1036	1056		1076		1096		1116	
7 th	1037	1067		1077		1097		1117	
8 th	1038	1068		1078		1098		1118	
9 th	1039	1069		1079		1099		1119	
10 th	1040	1060		1080		1100		1120	

0 = STANDARD ATMOSPHERE
1 = STD + ΔT_{IN}
2 = ARBITRARY θ (h)

C_{LW}
(NOTE c)

Δt_L (HR)		t_L (HR)		N_{PSD} LOITER		N_{PSD} LOITER		ΔF_{eL} (FT ²)	
LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
1041		1081		1101		1121		1131	
1042		1082		1102		1122		1132	
1043		1083		1103		1123		1133	
1044		1084		1104		1124		1134	
1045		1085		1105		1125		1135	
1046		1086		1106		1126		1136	
1047		1087		1107		1127		1137	
1048		1088		1108		1128		1138	
1049		1089		1109		1129		1139	
1050		1090		1110		1130		1140	

NOTES: a. INPUT NOT NECESSARY WHEN ATMIND = 0, 2
b. INPUT NOT NECESSARY WHEN AUXIND = 1, 2
c. INPUT NOT NECESSARY WHEN AUXIND = 1, 3

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CHANGE IN FUEL WEIGHT (SGTIND = 7)

CHANGE IN PAYLOAD WEIGHT (SGTIND = 8)

ΔW_F (LBS)		t_{FW} (HR)		ΔW_{PL} (LBS)		t_{PW} (HR)	
LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
1 st	1141	1151		1161		1171	
2 nd	1142	1152		1162		1172	
3 rd	1143	1153		1163		1173	
4 th	1144	1154		1164		1174	
5 th	1145	1155		1165		1175	
6 th	1146	1156		1166		1176	
7 th	1147	1157		1167		1177	
8 th	1148	1158		1168		1178	
9 th	1149	1159		1169		1179	
10 th	1150	1160		1170		1180	

TRANSFER ALTITUDE (SGTIND = 9)

h_{FINAL} (FT)

$h_{OPT-IND = 0}$

OR

h_{MAX} (FT)

$h_{OPT-IND = 1}$

CHANGE FUEL OR CHANGE PAYLOAD

SGTIND = 7

OR

SGTIND = 8

WGTIND

LOC

VALUE

1181

0 = $W + \Delta W \leq W_G$

1 = NO WEIGHT RESTRICTION

LOC	VALUE
1 st	1181
2 nd	1182
3 rd	1183
4 th	1184
5 th	1185
6 th	1186
7 th	1187
8 th	1188
9 th	1189
10 th	1190

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ENGINE CYCLE DATA; NON-STANDARD PERFORMANCE

SHEET NO.	CASE NO.
OF	

PRIMARY ENGINE DATA

VARIABLE	LOC	VALUE
WDTIND	1201	
N1IND	1202	
N10IND	1203	
N2IND	1204	
QIND	1205	

WDTIND: { 0 = NO FUEL FLOW CUTOFF
1 = FUEL FLOW CUTOFF

N1IND: { 0 = NO N1 CUTOFF
1 = N1 CUTOFF

VARIABLE	LOC	VALUE
$\dot{W}_{MAX}/W^*$	1220	
$N_{I MAX}/N_{I}^*$	1221	
$(N_I/\sqrt{Q}/N_I^*)_{MAX}$	1222	
$N_{II MAX}/N_{II}^*$	1223	
Q_{MAX}/Q^*	1224	

INPUT IF WDTIND = 1

INPUT IF N1IND = 1

INPUT IF N10IND = 1

INPUT IF N2IND = 1,2

INPUT IF QIND = 1

N2IND: { 0 = NO N2 CUTOFF; OPTIMUM N2 VARIATION
1 = N2 CUTOFF; OPTIMUM N2 VARIATION
2 = N2 CUTOFF; NON-OPTIMUM N2 VARIATION

QIND: { 0 = NO TORQUE CUTOFF
1 = TORQUE CUTOFF

N10IND: { 0 = NO REFERRED N1 CUTOFF
1 = REFERRED N1 CUTOFF

VARIABLE	LOC	VALUE
RNOIND	1206	

0 = NO REYNOLDS NO. CORRECTIONS
1 = REYNOLDS NO. CORRECTIONS

REYNOLDS NO. CORRECTION FACTOR

VALUES OF $\frac{N_I}{N_I^*} \frac{D}{V_1}$

LOC	VALUE
1207	
1208	
1209	
1210	
1211	
1212	
1213	
1214	
1215	
1216	

VALUES OF K_{PR}

LOC	VALUE
1225	
1226	
1227	
1228	
1229	
1230	
1231	
1232	
1233	
1234	

INPUT THIS TABLE IF RNOIND = 1

OUTPUT SHAFT SPEED CORRECTION

VALUES OF $\frac{N_{II}}{N_{II OPT}}$

LOC	VALUE
1238	
1239	
1240	
1241	
1242	
1243	
1244	
1245	
1246	
1247	

VALUES OF K_{PN}

LOC	VALUE
1248	
1249	
1250	
1251	
1252	
1253	
1254	
1255	
1256	
1257	

INPUT THIS TABLE IF N2IND = 2 AND
NON-STANDARD CORRECTION IS DESIRED

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ENGINE CYCLE DATA; NON-STANDARD PERFORMANCE

AUXILIARY INDEPENDENT ENGINE DATA

VARIABLE	LOC	VALUE
WDTINDI	2201	
N1INDI	2202	
N1θINDI	2203	
N2INDI	2204	
QINDI	2205	

VARIABLE	LOC	VALUE
$\dot{W}_{MAX}/\dot{W}^*$	2220	
N_{IMAX}/N_1^*	2221	
$(N_1/\sqrt{\theta_1}/N_1^*)_{MAX}$	2222	
N_{IIMAX}/N_{II}^*	2223	
Q_{MAX}/Q^*	2224	

INPUT IF WDTINDI = 1

INPUT IF N1INDI = 1

INPUT IF N1θINDI = 1

INPUT IF N2INDI = 1, 2

INPUT IF QINDI = 1

WDTINDI: { 0 = NO FUEL FLOW CUTOFF
1 = FUEL FLOW CUTOFF

N1INDI: { 0 = NO N1 CUTOFF
1 = N1 CUTOFF

N2INDI: { 0 = NO N2 CUTOFF; OPTIMUM N2 VARIATION
1 = N2 CUTOFF; OPTIMUM N2 VARIATION
2 = N2 CUTOFF; NON-OPTIMUM N2 VARIATION

QINDI: { 0 = NO TORQUE CUTOFF
1 = TORQUE CUTOFF

N1θINDI: { 0 = NO REFERRED N1 CUTOFF
1 = REFERRED N1 CUTOFF

VARIABLE	LOC	VALUE
RNOINDI	2206	

0 = NO REYNOLDS NO. CORRECTIONS
1 = REYNOLDS NO. CORRECTIONS

REYNOLDS NO. CORRECTION FACTOR

VALUES OF $\frac{N_1}{N_1^*} \frac{D}{D_1}$

LOC	VALUE
2207	
2208	
2209	
2210	
2211	
2212	
2213	
2214	
2215	
2216	

VALUES OF K_{PR}

LOC	VALUE
2225	
2226	
2227	
2228	
2229	
2230	
2231	
2232	
2233	
2234	

INPUT THIS TABLE IF RNOINDI = 1

OUTPUT SHAFT SPEED CORRECTION

VALUES OF $\frac{N_{II}}{N_{II OPT}}$

LOC	VALUE
2238	
2239	
2240	
2241	
2242	
2243	
2244	
2245	
2246	
2247	

VALUES OF K_{PN}

LOC	VALUE
2248	
2249	
2250	
2251	
2252	
2253	
2254	
2255	
2256	
2257	

INPUT THIS TABLE IF N2INDI = 2 AND
NON-STANDARD CORRECTION IS DESIRED

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SHORT FORM AERO (MAIN) ROTOR CYCLE INFORMATION

VARIABLE	LOC	VALUE
ROTOR CYCLE NO.	1601	
θ TWIST REF	1602	

VARIABLE	LOC	VALUE
$C_{D_{B_0}}$	1603	
K_{H_1}	1604	
K_{H_2}	1605	
K_{H_3}	1606	
K_{H_4}	1607	
$M_{D_{B_0}}$	1608	

HOVER
(STATIC
THRUST
DATA)

VARIABLE	LOC	VALUE
C_{D_B}	1609	
K_{C_1}	1610	
K_{C_2}	1611	
K_{C_3}	1612	
K_{C_4}	1613	
K_{C_5}	1614	
M_{D_0}	1615	

CRUISE
DATA

NO. OF CT	LOC	VALUE	VALUES OF C_T		VALUES OF K_{HOVA}	
			LOC	VALUE	LOC	VALUE
1	1617		1627			
2	1618		1628			
3	1619		1629			
4	1620		1630			
5	1621		1631			
6	1622		1632			
7	1623		1633			
8	1624		1634			
9	1625		1635			
10	1626		1636			

HOVER
(STATIC
THRUST
DATA)

THIS INFORMATION IS FOR REFERENCE PURPOSES ONLY

REFERENCE BLADE NO.	
BLADE PLANFORM TAPER RATIO	$\lambda = C_{Tip}/C_{C. LINE ROT.}$
BLADE AIRFOIL DISTRIBUTION	
RADIAL STA (r/R)	AIRFOIL DESIGNATION
REMARKS	

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ROTOR PERFORMANCE MAP

	LOC	VALUE
ROTOR MAP NO.	2700	

	LOC	VALUE
p_{REF}	2701	
σ_{REF}	2702	
θ_{TREF}	2703	

HOVER PERFORMANCE

	LOC	VALUE
NO. OF C_T/σ 'S	2704	

	LOC	VALUE
NO. OF M_{TIP} 'S	2715	

VALUES OF C_T/σ

	LOC	VALUE
$(C_T/\sigma)_1$	2705	
$(C_T/\sigma)_2$	2706	
$(C_T/\sigma)_3$	2707	
$(C_T/\sigma)_4$	2708	
$(C_T/\sigma)_5$	2709	
$(C_T/\sigma)_6$	2710	
$(C_T/\sigma)_7$	2711	
$(C_T/\sigma)_8$	2712	
$(C_T/\sigma)_9$	2713	
$(C_T/\sigma)_{10}$	2714	

VALUES OF M_{TIP}

	LOC	VALUE
M_{T1}	2716	
M_{T2}	2717	
M_{T3}	2718	
M_{T4}	2719	
M_{T5}	2720	
M_{T6}	2721	

$$\left\{ \begin{array}{l} C_T/\sigma = 4T/\rho \pi D^2 V_T^2 N_R \sigma \\ C_{PH}/\sigma = 2200 \text{ RHP}/\rho \pi D^2 V_T^3 N_R \sigma \end{array} \right.$$

INPUT VALUES OF C_{PH}/σ FOR COMBINATIONS
OF C_T/σ AND M_{TIP}

	$M_{TIP1} =$		$M_{TIP2} =$		$M_{TIP3} =$		$M_{TIP4} =$		$M_{TIP5} =$		$M_{TIP6} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T/\sigma)_1 =$	2722		2732		2742		2752		2762		2772	
$(C_T/\sigma)_2 =$	2723		2733		2743		2753		2763		2773	
$(C_T/\sigma)_3 =$	2724		2734		2744		2754		2764		2774	
$(C_T/\sigma)_4 =$	2725		2735		2745		2755		2765		2775	
$(C_T/\sigma)_5 =$	2726		2736		2746		2756		2766		2776	
$(C_T/\sigma)_6 =$	2727		2737		2747		2757		2767		2777	
$(C_T/\sigma)_7 =$	2728		2738		2748		2758		2768		2778	
$(C_T/\sigma)_8 =$	2729		2739		2749		2759		2769		2779	
$(C_T/\sigma)_9 =$	2730		2740		2750		2760		2770		2780	
$(C_T/\sigma)_{10} =$	2731		2741		2751		2761		2771		2781	

NOTE: C_T & C_P ARE IN "ROTOR" NOTATION

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ROTOR PERFORMANCE "MAP"

HESCOMP HELICOPTER SIZING AND PERFORMANCE
COMPUTER PROGRAM B-91

CRUISE PERFORMANCE

NO. OF ADVANCE RATIOS (μ)	LOC	VALUE
	2782	

NO. OF ROTOR LIFT COEFFICIENTS (C_T'/σ)	LOC	VALUE
	2789	

NO. OF ROTOR PROPULSIVE THRUST COEFF (C_X'/σ)	LOC	VALUE
	2800	

VALUES OF μ

	LOC	VALUE
μ_1	2783	
μ_2	2784	
μ_3	2785	
μ_4	2786	
μ_5	2787	
μ_6	2788	

VALUES OF (C_T'/σ)

	LOC	VALUE
$(C_T'/\sigma)_1$	2790	
$(C_T'/\sigma)_2$	2791	
$(C_T'/\sigma)_3$	2792	
$(C_T'/\sigma)_4$	2793	
$(C_T'/\sigma)_5$	2794	
$(C_T'/\sigma)_6$	2795	
$(C_T'/\sigma)_7$	2796	
$(C_T'/\sigma)_8$	2797	
$(C_T'/\sigma)_9$	2798	
$(C_T'/\sigma)_{10}$	2799	

VALUES OF C_X'/σ

	LOC	VALUE
$(C_X'/\sigma)_1$	2801	
$(C_X'/\sigma)_2$	2802	
$(C_X'/\sigma)_3$	2803	
$(C_X'/\sigma)_4$	2804	
$(C_X'/\sigma)_5$	2805	
$(C_X'/\sigma)_6$	2806	
$(C_X'/\sigma)_7$	2807	
$(C_X'/\sigma)_8$	2808	
$(C_X'/\sigma)_9$	2809	
$(C_X'/\sigma)_{10}$	2810	

$$\mu = \frac{1.688 V_{KT}}{V_{TIP}}$$

$$C_T'/\sigma = \frac{LIFT}{\frac{\rho \pi D^2 V_T^2 \sigma_{MR} N_R}{4}}$$

$$C_X'/\sigma = \frac{PROPULSIVE FORCE}{\frac{\rho \pi D^2 V_T^2 \sigma_{MR} N_R}{4}}$$

$$C_P/\sigma = \frac{2200 \times ROTOR POWER (SHP)}{\rho \pi D^2 V_T^3 \sigma_{MR} N_R}$$

HESCOMP HELICOPTER SIZING AND PERFORMANCE
COMPUTER PROGRAM B-91

BOEING VERTOL COMPANY
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ROTOR PERFORMANCE "MAP"

CRUISE PERFORMANCE INPUT VALUES OF ROTOR POWER COEFFICIENT (C_P) FOR COMBINATIONS OF $\mu, \frac{C_T}{\sigma}, \frac{C_X}{\sigma}$

ROTOR PROPULSIVE FORCE COEFFICIENT

ADVANCE RATIO $\mu_1 =$	$(C_X/\sigma)_1 =$		$(C_X/\sigma)_2 =$		$(C_X/\sigma)_3 =$		$(C_X/\sigma)_4 =$		$(C_X/\sigma)_5 =$		$(C_X/\sigma)_6 =$		$(C_X/\sigma)_7 =$		$(C_X/\sigma)_8 =$		$(C_X/\sigma)_9 =$		$(C_X/\sigma)_{10} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T/\sigma)_1 =$	2811		2821		2831		2841		2851		2861		2871		2881		2891		2901	
$(C_T/\sigma)_2 =$	2812		2822		2832		2842		2852		2862		2872		2882		2892		2902	
$(C_T/\sigma)_3 =$	2813		2823		2833		2843		2853		2863		2873		2883		2893		2903	
$(C_T/\sigma)_4 =$	2814		2824		2834		2844		2854		2864		2874		2884		2894		2904	
$(C_T/\sigma)_5 =$	2815		2825		2835		2845		2855		2865		2875		2885		2895		2905	
$(C_T/\sigma)_6 =$	2816		2826		2836		2846		2856		2866		2876		2886		2896		2906	
$(C_T/\sigma)_7 =$	2817		2827		2837		2847		2857		2867		2877		2887		2897		2907	
$(C_T/\sigma)_8 =$	2818		2828		2838		2848		2858		2868		2878		2888		2898		2908	
$(C_T/\sigma)_9 =$	2819		2829		2839		2849		2859		2869		2879		2889		2899		2909	
$(C_T/\sigma)_{10} =$	2820		2830		2840		2850		2860		2870		2880		2890		2900		2910	
ADVANCE RATIO $\mu_2 =$	$(C_X/\sigma)_1 =$		$(C_X/\sigma)_2 =$		$(C_X/\sigma)_3 =$		$(C_X/\sigma)_4 =$		$(C_X/\sigma)_5 =$		$(C_X/\sigma)_6 =$		$(C_X/\sigma)_7 =$		$(C_X/\sigma)_8 =$		$(C_X/\sigma)_9 =$		$(C_X/\sigma)_{10} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T/\sigma)_1 =$	2911		2921		2931		2941		2951		2961		2971		2981		2991		3001	
$(C_T/\sigma)_2 =$	2912		2922		2932		2942		2952		2962		2972		2982		2992		3002	
$(C_T/\sigma)_3 =$	2913		2923		2933		2943		2953		2963		2973		2983		2993		3003	
$(C_T/\sigma)_4 =$	2914		2924		2934		2944		2954		2964		2974		2984		2994		3004	
$(C_T/\sigma)_5 =$	2915		2925		2935		2945		2955		2965		2975		2985		2995		3005	
$(C_T/\sigma)_6 =$	2916		2926		2936		2946		2956		2966		2976		2986		2996		3006	
$(C_T/\sigma)_7 =$	2917		2927		2937		2947		2957		2967		2977		2987		2997		3007	
$(C_T/\sigma)_8 =$	2918		2928		2938		2948		2958		2968		2978		2988		2998		3008	
$(C_T/\sigma)_9 =$	2919		2929		2939		2949		2959		2969		2979		2989		2999		3009	
$(C_T/\sigma)_{10} =$	2920		2930		2940		2950		2960		2970		2980		2990		3000		3010	

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SHEET NO. 4 CASE NO.
OF 5

ROTOR PERFORMANCE "MAP"

CRUISE PERFORMANCE INPUT VALUES OF ROTOR POWER COEFFICIENT (C_P) FOR COMBINATIONS OF μ , C_T & C_X
 $\frac{\sigma}{\sigma}$

ROTOR PROPULSIVE FORCE COEFFICIENT

ADVANCE RATIO $\mu_3 =$	$(C_X'/\sigma)_1 =$		$(C_X'/\sigma)_2 =$		$(C_X'/\sigma)_3 =$		$(C_X'/\sigma)_4 =$		$(C_X'/\sigma)_5 =$		$(C_X'/\sigma)_6 =$		$(C_X'/\sigma)_7 =$		$(C_X'/\sigma)_8 =$		$(C_X'/\sigma)_9 =$		$(C_X'/\sigma)_{10} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T'/\sigma)_1 =$	3011		3021		3031		3041		3051		3061		3071		3081		3091		3101	
$(C_T'/\sigma)_2 =$	3012		3022		3032		3042		3052		3062		3072		3082		3092		3102	
$(C_T'/\sigma)_3 =$	3013		3023		3033		3043		3053		3063		3073		3083		3093		3103	
$(C_T'/\sigma)_4 =$	3014		3024		3034		3044		3054		3064		3074		3084		3094		3104	
$(C_T'/\sigma)_5 =$	3015		3025		3035		3045		3055		3065		3075		3085		3095		3105	
$(C_T'/\sigma)_6 =$	3016		3026		3036		3046		3056		3066		3076		3086		3096		3106	
$(C_T'/\sigma)_7 =$	3017		3027		3037		3047		3057		3067		3077		3087		3097		3107	
$(C_T'/\sigma)_8 =$	3018		3028		3038		3048		3058		3068		3078		3088		3098		3108	
$(C_T'/\sigma)_9 =$	3019		3029		3039		3049		3059		3069		3079		3089		3099		3109	
$(C_T'/\sigma)_{10} =$	3020		3030		3040		3050		3060		3070		3080		3090		3100		3110	
ADVANCE RATIO $\mu_4 =$	$(C_X'/\sigma)_1 =$		$(C_X'/\sigma)_2 =$		$(C_X'/\sigma)_3 =$		$(C_X'/\sigma)_4 =$		$(C_X'/\sigma)_5 =$		$(C_X'/\sigma)_6 =$		$(C_X'/\sigma)_7 =$		$(C_X'/\sigma)_8 =$		$(C_X'/\sigma)_9 =$		$(C_X'/\sigma)_{10} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T'/\sigma)_1 =$	3111		3121		3131		3141		3151		3161		3171		3181		3191		3201	
$(C_T'/\sigma)_2 =$	3112		3122		3132		3142		3152		3162		3172		3182		3192		3202	
$(C_T'/\sigma)_3 =$	3113		3123		3133		3143		3153		3163		3173		3183		3193		3203	
$(C_T'/\sigma)_4 =$	3114		3124		3134		3144		3154		3164		3174		3184		3194		3204	
$(C_T'/\sigma)_5 =$	3115		3125		3135		3145		3155		3165		3175		3185		3195		3205	
$(C_T'/\sigma)_6 =$	3116		3126		3136		3146		3156		3166		3176		3186		3196		3206	
$(C_T'/\sigma)_7 =$	3117		3127		3137		3147		3157		3167		3177		3187		3197		3207	
$(C_T'/\sigma)_8 =$	3118		3128		3138		3148		3158		3168		3178		3188		3198		3208	
$(C_T'/\sigma)_9 =$	3119		3129		3139		3149		3159		3169		3179		3189		3199		3209	
$(C_T'/\sigma)_{10} =$	3120		3130		3140		3150		3160		3170		3180		3190		3200		3210	

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BOEING VERTOL COMPANY
A DIVISION OF THE BOEING COMPANY
ROTOR PERFORMANCE "MAP"

CRUISE PERFORMANCE INPUT VALUES OF ROTOR POWER COEFFICIENT (C_p) FOR COMBINATIONS OF $\mu, \frac{C_T}{\sigma}$ & $\frac{C_X}{\sigma}$

ROTOR PROPULSIVE FORCE COEFFICIENT

ADVANCE RATIO μ_5	$(C_X/\sigma)_1 =$		$(C_X/\sigma)_2 =$		$(C_X/\sigma)_3 =$		$(C_X/\sigma)_4 =$		$(C_X/\sigma)_5 =$		$(C_X/\sigma)_6 =$		$(C_X/\sigma)_7 =$		$(C_X/\sigma)_8 =$		$(C_X/\sigma)_9 =$		$(C_X/\sigma)_{10} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T/\sigma)_1 =$	3211		3221		3231		3241		3251		3261		3271		3281		3291		3301	
$(C_T/\sigma)_2 =$	3212		3222		3232		3242		3252		3262		3272		3282		3292		3302	
$(C_T/\sigma)_3 =$	3213		3223		3233		3243		3253		3263		3273		3283		3293		3303	
$(C_T/\sigma)_4 =$	3214		3224		3234		3244		3254		3264		3274		3284		3294		3304	
$(C_T/\sigma)_5 =$	3215		3225		3235		3245		3255		3265		3275		3285		3295		3305	
$(C_T/\sigma)_6 =$	3216		3226		3236		3246		3256		3266		3276		3286		3296		3306	
$(C_T/\sigma)_7 =$	3217		3227		3237		3247		3257		3267		3277		3287		3297		3307	
$(C_T/\sigma)_8 =$	3218		3228		3238		3248		3258		3268		3278		3288		3298		3308	
$(C_T/\sigma)_9 =$	3219		3229		3239		3249		3259		3269		3279		3289		3299		3309	
$(C_T/\sigma)_{10} =$	3220		3230		3240		3250		3260		3270		3280		3290		3300		3310	
ADVANCE RATIO μ_6	$(C_X/\sigma)_1 =$		$(C_X/\sigma)_2 =$		$(C_X/\sigma)_3 =$		$(C_X/\sigma)_4 =$		$(C_X/\sigma)_5 =$		$(C_X/\sigma)_6 =$		$(C_X/\sigma)_7 =$		$(C_X/\sigma)_8 =$		$(C_X/\sigma)_9 =$		$(C_X/\sigma)_{10} =$	
	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE	LOC	VALUE
$(C_T/\sigma)_1 =$	3311		3321		3331		3341		3351		3361		3371		3381		3391		3401	
$(C_T/\sigma)_2 =$	3312		3322		3332		3342		3352		3362		3372		3382		3392		3402	
$(C_T/\sigma)_3 =$	3313		3323		3333		3343		3353		3363		3373		3383		3393		3403	
$(C_T/\sigma)_4 =$	3314		3324		3334		3344		3354		3364		3374		3384		3394		3404	
$(C_T/\sigma)_5 =$	3315		3325		3335		3345		3355		3365		3375		3385		3395		3405	
$(C_T/\sigma)_6 =$	3316		3326		3336		3346		3356		3366		3376		3386		3396		3406	
$(C_T/\sigma)_7 =$	3317		3327		3337		3347		3357		3367		3377		3387		3397		3407	
$(C_T/\sigma)_8 =$	3318		3328		3338		3348		3358		3368		3378		3388		3398		3408	
$(C_T/\sigma)_9 =$	3319		3329		3339		3349		3359		3369		3379		3389		3399		3409	
$(C_T/\sigma)_{10} =$	3320		3330		3340		3350		3360		3370		3380		3390		3400		3410	

5.3 PROGRAM INPUT

5.3.1 Program Variables

AF	Activity factor (per blade) of propeller
AR	Wing aspect ratio
AR _{AP}	Aft rotor pylon aspect ratio (tanden rotor helicopter)
AR _{FP}	Forward rotor pylon aspect ratio (tandem rotor helicopter)
AR _{HT}	Horizontal tail aspect ratio
AR _{VT}	Vertical tail aspect ratio
b (b _{MR} , b _{TR})	Blade number per rotor or propeller
b _{BS} /d _{NI}	Ratio of auxiliary independent engine nacelle strut span to nacelle diameter
b _W /D	Ratio of wing span to main rotor diameter
$\bar{C}$	Tail rotor/fin blockage factor
C _{DB}	Baseline rotor cruise profile drag coefficient (input in rotor "cycle")
C _{D_{BO}}	Baseline rotor hover profile drag coefficient (input in rotor "cycle")
C _{D_{AP}}	Aft rotor pylon profile drag coefficient at $R_e=10^7$ (based on aft pylon planform area)
C _{D_{FP}}	Forward rotor pylon profile drag coefficient at $R_e=10^7$ (based on forward pylon max frontal area)
C _{D_{CSMR}}	Main rotor hub center section profile drag coefficient (based on center section frontal area)
C _{D_{CSTR}}	Tail rotor hub center section profile drag coefficient (based on center section frontal area)
C _{D_{HT}}	Profile drag coefficient of horizontal tail at $R_e=10^7$ (based on horizontal tail planform area)

C_{DN}	Profile drag coefficient of primary engine nacelles at $R_e=10^7$ (based on wetted area of all nacelles)
C_{DNI}	Profile drag coefficient of auxiliary independent engine nacelle at $R_e=10^7$ (based on wetted area of all nacelles)
C_{DNS}	Profile drag coefficient of auxiliary independent engine nacelle strut at $R_e=10^7$ (based on wetted area of strut(s))
C_{DSHMR}	Main rotor hub shank profile drag coefficient (based on shank frontal area)
C_{DSHTR}	Tail rotor hub shank profile drag coefficient (based on shank frontal area)
C_{DVT}	Profile drag coefficient of vertical tail at $R_e=10^7$ (based on vertical tail planform area)
C_{DWi}	Profile drag coefficient of wing at $R_e=10^7$ (based on wing planform area)
C_F/C	Ratio of download alleviating flap chord to wing chord
$\Delta C.G.$	Helicopter cg travel (ft)
$\Delta C.G._R$	Location of main rotor in the longitudinal direction relative to the aircraft operating weight empty (OWE) cg position (ft)
C_{LW}	Wing lift coefficient
C_{LD}	Wing design lift coefficient
C_{LDP}	Wing operating lift coefficient at cruise condition for engine sizing
C_{LDES}	Tail fin design lift coefficient
C_{LFIN}	Tail fin cruise lift coefficient
C_{Li}	Propeller integrated design lift coefficient
$C_{l\alpha}$	Two-dimensional wing lift coefficient slope (Rad. $^{-1}$)

C_P	Propeller power coefficient ($550 \text{ HP}/\rho \pi^3 D^5$)
C_{PH}/σ	Ratio of rotor power coefficient to rotor solidity ($C_{PH}/\sigma = 550 \text{ RHP}/\rho AV^3 \sigma$)
C_T	Propeller thrust coefficient ($\text{Thrust}/\rho \pi D^4$)
C_{TH}	Rotor thrust coefficient ($\text{Thrust}/\rho AV_{TIP}^2$)
(C_T/σ)	Ratio of thrust coefficient to rotor solidity (helicopter $C_T = \text{Thrust}/\rho AV_{TIP}^2$), includes (C_T/σ) , $(C_T/\sigma)_{Des (H)}$, $(C_T/\sigma)_{CR}$
$(C_{TG}/C_{T_{NET}})$	Ratio of tail rotor total thrust coefficient to net thrust coefficient, where $C_{T_{NET}} = C_{TG} - \text{Fin blocking losses}$
C_X/σ	Rotor propulsive thrust coefficient divided by main rotor solidity. Used in defining rotor limits $C_X/\sigma = \frac{\text{Thrust required}}{\rho \pi D_{MR}^2 N_R V_{TIP}^2 \sigma_{MR}}$ 4
D_{MR}	Main rotor diameter (feet)
D_{TR}	Tail rotor diameter (feet)
D_{PROP}	Propeller diameter (feet)
d_i	Position of inboard underwing store (fraction of wing semi-span)
d_o	Position of outboard underwing store (fraction of wing semi-span)
$d_{TB}/d_{\overline{TB}}$	Ratio of average tail boom tip diameter to average tail boom diameter
EAS	Equivalent airspeed required during climb and/or descent
e	Span loading efficiency factor
ΔF_e	Increment in equivalent flat plate area parasite drag of fuselage (ft^2)

F_N^*	Auxiliary independent engine maximum static thrust at seal level, standard conditions (total thrust for all engines)
ΔF_{eCL}	Increment in equivalent flat plate area parasite drag (climb performance segment) - ft^2
ΔF_{eCR}	Increment in equivalent flat plate area parasite (cruise performance segment) - ft^2
ΔF_{eDSC}	Increment in equivalent flat plate area parasite drag (descent performance segment) - ft^2
$\Delta F_{eLOITER}$	Increment in equivalent flat plate area parasite drag (loiter performance segment) - ft^2
$F_N/\delta F_N^*$	Referred thrust for turbojet/fan engine cycles
g_{RQMT}	Total maneuver g requirement helicopter must satisfy (wing + rotor) - g
g_{ROT}	Maneuver g's which rotor must carry. In the case of a pure helicopter, $g_{RQMT} = g_{ROT}$
g/S	Tandem rotor gap/stagger ratio
g_{MR}/TR	Gap between tail rotor disc and main rotor disc (ft)
(GW/F_e)	Ratio of configuration design gross weight to equivalent flat plate area parasite drag $(lb/ft)^2$
h'/h_F	Ratio of wing height on fuselage (relative to the bottom of the fuselage), h' , to the total fuselage height, h_F .
HEADWIND	Headwind during cruise (knots)
h_O	Initial altitude at start of mission (ft)
$h_C(C)$	Cruise altitude for sizing main rotor solidity (ft)
h_C	Cruise altitude for sizing primary engines (ft)
$h_{BF/D}$	Ratio of fuselage bottom height above ground to main rotor diameter
h_{FINAL}	Final altitude for transfer altitude segment (SGTIND=9)

h_{P_2}	Aft rotor pylon height (ft)
h_{P_1}	Forward rotor pylon height (ft)
h_{TO}	Takeoff altitude for sizing engines (ft)
h_{max}	Maximum altitude during climb (ft) or during transfer altitude (ft)
h_{min}	Minimum altitude during descent (ft)
Δh	Step size for climb or descent (ft)
h_F	Height of fuselage (ft)
J	Propeller advance ratio, $J = V/nD$
k_3	Primary engine weight factors
k_4	Primary engine weight factors
k_{amd}	Main rotor weight factor
k_{AEI}	Auxiliary engine installation weight factor
k_B	Body group weight factor
k_{CC}	Cockpit controls weight factor
k_{FS}	Fuel system weight factor
k_{FW}	Fixed wing controls weight factor
k_{LG}	Landing gear weight factor
k_{MC}	Miscellaneous controls weight factor
k_{MG}	Main landing gear weight factor
k_{PEI}	Primary engine installation weight factor
k_{RC}	Main rotor controls weight factor
k_{RCA}	Auxiliary rotor controls weight factor
k_{SAS}	Stability Augmentation System (SAS) weight factor

k_{SC}	Main rotor system controls weight factor
k_{SCA}	Auxiliary rotor system controls weight factor
k_{TM}	Tilt mechanism weight factor
$k_{T.STING}$	Tail boom (on single rotor helicopter) length extending aft of tail rotor center as a fraction of tail rotor radius
k_{WW}	Detailed wing weight factor
k_{ZZZ}	Single rotor helicopter yaw moment of inertia adjustment factor
K_1	Main rotor controls weight factor
K_2	Main rotor system controls weight multiplicative factor
K_3	Fixed wing controls weight multiplicative factor
K_4	Auxiliary rotor controls weight multiplicative factor
K_5	Auxiliary rotor system controls weight multiplicative factor
K_6	Body weight multiplicative factor
K_7	Landing gear weight multiplicative factor
K_8	Wing weight multiplicative factor
K_9	Horizontal tail weight multiplicative factor
K_{10}	Primary nacelle weight multiplicative factor
K_{11}	Auxiliary nacelle weight multiplicative factor
K_{12}	Primary rotor blade weight multiplicative factor
K_{13}	Primary rotor hub weight multiplicative factor
K_{14}	Tail rotor weight multiplicative factor
K_{15}	Auxiliary rotor weight multiplicative factor
K_{16}	Primary drive system weight multiplicative factor

K_{17}	Auxiliary drive system weight multiplicative factor
K_{18}	Primary engine weight multiplicative factor
K_{19}	Auxiliary engine weight multiplicative factor
K_{20}	Tail rotor drive system weight multiplicative factor
k_{AIA}	Auxiliary air induction system weight factor
k_{AIP}	Primary air induction system weight factor
k_{ADS}	Auxiliary drive system weight factor
k_{ADSZ}	Auxiliary drive system weight factor (number of gears in system)
$K_{ALT. \text{ PAYL.}}$	Ratio of alternate payload increment to design payload (used in XMSN sizing)
K_{AP}	Aft rotor pylon multiplicative drag factor
k_{AR}	Auxiliary rotor weight factor
k_{BLFD}	Blade fold weight factor
k_{CLF}	Crash load factor
K_{FP}	Forward rotor pylon multiplicative drag factor
K_{C1}	Rotor retreating blade stall profile drag parameters (input in rotor "cycle")
K_{C2}	Rotor retreating blade stall profile drag parameters (input in rotor "cycle")
K_{C3}	Rotor advancing tip mach number compressibility drag parameters (input in rotor "cycle")
K_{C4}	
K_{C5}	
K_F	Fuselage multiplicative drag factor

K_{FED}	Trend constant input with GW/Fe when DRGIND = 2
K_{FI}	Auxiliary independent engines fuel flow multiplicative factor (used in TAXI)
K_{FF}	Fuel flow multiplicative factor
K_{FULS}	Fraction of fixed useful load located in cockpit area
K_{H1}	Rotor (hover) blade profile drag parameter (input in rotor "cycle")
K_{H2}	Rotor blade (hover) compressibility drag parameter (input in rotor "cycle")
K_{H3}	Rotor blade (hover) compressibility drag parameter (input in rotor "cycle")
K_{H4}	Rotor blade (hover) drag divergence Mach number parameter (input in rotor "cycle")
K_{HOV_A}	Rotor hover induced power factor (input in rotor "cycle")
K_{HT}	Horizontal tail multiplicative drag factor
K_N	Primary nacelle multiplicative drag factor
$K_{P_{CLIMB}}$	Helicopter forward flight climb efficiency
k_{NAC}	Primary cowling weight factor (psf)
k_{NACA}	Auxiliary cowling weight factor (psf)
K_{NI}	Auxiliary independent engine nacelle multiplicative drag factor
k_{NS}	Nacelle strut weight factor
k_{PA}	Auxiliary rotor weight factor
k_{PEI}	Primary engine installation weight factor
$K_{P_{DESCENT}}$	Main rotor descent efficiency
k_{PDS}	Primary drive system weight factor
k_{PDSZ}	Primary drive system weight factor (number of gears in system)

k_{PH}	Primary hub weight factor
K_{PN}	Ratio of power available at specified N_{II} to power available at optimum N_{II}
K_{PR}	Correction factor for engine power to account for Reynolds number effects
k_{PRB}	Primary rotor blade weight factor
K_{HPIM}	Main rotor hub/shank multiplicative drag factor
K_{HPIT}	Tail rotor hub/shank multiplicative drag factor
k_{RBF}	Primary rotor blade weight factor
K_{TBBS}	Weight of tail boom as a fraction of total fuselage weight
k_{TR}	Tail rotor weight factor
k_{TRDS}	Tail rotor drive system weight factor
k_{TRS}	Tail rotor solidity multiplicative factor (used to determine tail rotor solidity)
K_{VT}	Vertical tail multiplicative drag factor
k	Auxiliary rotor weight factor
K_W	Wing multiplicative drag factor
k_{WP}	Wing weight/area factor (psf)
k_{WS}	Wing stores only weight trend factor
K_Z	Vertical tail fin height factor
LF	Wing unload factor
ℓ_{AIA}/ℓ_{e_A}	Ratio of air induction system length to engine length (auxiliary independent engines)
ℓ_{AIP}/ℓ_e	Ratio of air induction system length to engine length (primary engines)
$\ell_{CONST DIA}(\ell_c)$	Constant diameter section (cabin) length (ft)

ℓ_{RW}	Length of ramp well (ft)
ℓ_{TH}'	Ratio of horizontal tail moment arm to main rotor radius
$(\ell/d)_P$	Fineness ratio of aircraft nose section
$(\ell/d)_T$	Fineness ratio of aircraft tail section
(ℓ_{TB}/d_{TB})	Fineness ratio of tail boom (single rotor helicopter)
MACH	Mach number required during climb and/or descent
M_{LF}	Maneuver load factor (g's)
M_{MO}	Maximum operating Mach number
M_{DO}	Baseline rotor advancing tip compressibility drag rise Mach number (input in rotor "cycle")
M_{DBO}	Baseline rotor hover compressibility drag rise (lift=0) Mach No. (input in rotor "cycle")
M_{TIP}	Rotor hover tip Mach No.
N	Rotor loading (rotor lift/GW)
N_P	Number of primary engines
N_{PI}	Number of auxiliary independent engines
N_{PSD}	Number of primary engines inoperative (for engine sizing)
$N_{PSD}()$	Number of primary engines shut down during cruise, loiter, climb or descent
$(N_{PSD})_C$	Number of primary engines shut down during cruise (for engine sizing)
$N_{PSD_i}()$	Number of auxiliary independent engines shut down during cruise, loiter, climb or descent
N_{PROP}	Number of propellers
N_R	Number of rotors
N_{IMAX}/N_I^*	Gas generator RPM limit - ratio of max gas generator RPM to RPM at maximum static power, sea level, standard

$N_I/N_I^*)_{MAX}$	Gas generator referred RPM limit (θ = temperature ratio @ compression face), this input simulates a restriction on compression speed
$N_{II}/N_{II_{OPT}}$	Ratio of operating power turbine speed to optimum power turbine speed (input when $N2IND = 2$)
(N_{II}/N_{IIMAX})	Ratio of operating power turbine speed to maximum power turbine speed (input for both primary and auxiliary independent engines in performance segments 1 - 6)
(N_{IIMAX}/N_{II}^*)	Power turbine speed limit ratio of maximum power turbine speed to power turbine speed at maximum static power, sea level, standard
$(N_{II}/N_{IIMAX})_c$	Ratio of operating power turbine speed to maximum power turbine speed (input when sizing primary engines for takeoff)
$(N_{II}/N_{IIMAX})_i$	Ratio of operating power turbine speed to maximum power, turbine speed (input when sizing auxiliary independent engines for cruise)
$(N_{II}/N_{IIMAX})_{TO}$	Ratio of operating power turbine speed to maximum power turbine speed (input when sizing primary engines for cruise)
$((O/L)/D)$	Tandem rotor overlap/main rotor diameter ratio
OWE	Operating weight empty (pounds)
PEHF	Primary engine power fraction. Required when $TOLIND = 2$ and 4
Q_{MAX}/Q^*	Ratio of maximum torque limit to torque developed at sea level static standard day conditions
R_O	Initial range at start of mission (nautical miles)
R_{max}	Range at end of cruise and/or descent
RM_I	Wing relief as percentage of GW
ΔR	Step size for cruise (nautical miles)
R/D	Rate of descent (fpm)
$(Re/\ell)_i$	Mean Reynolds number per foot for mission

S_{HT}	Area of horizontal tail. Used when HTIND = 1
S_W	Wing planform area (ft ²)
$\frac{SHP_{MRX}}{SHP^*_{MR}}$	Ratio of main rotor drive system XMSN rating to main rotor design power
$\frac{SHP_{TRX}}{TRP^*}$	Ratio of tail rotor drive system XMSN rating to tail rotor design power
$SHP/\delta\sqrt{\theta} \text{ SHP}^*$	Referred power for turboshaft engine cycles
$\frac{SHP_{AUX}}{SHP^*_{AUX}}$	Ratio of auxiliary propulsion drive system XMSN rating to auxiliary propeller design power
SHP^*_i	Auxiliary independent engine installed power (total for all engines)
SHP^*_P	Primary engine installed power (total for all engines)
SHP_{ACC}	Accessory power losses (SHP)
SHP_E/SHP^*	Ratio of design engine rating to maximum S.L. static (STD DAY) engine power
$\Delta S/S_{STR}$	Ratio of incremental auxiliary independent engine nacelle strut planform area to auxiliary independent engine nacelle strut planform area
ΔS_{WET}	Incremental wetted area of aircraft (ft ²)
$\Delta S_{WET}/S_F$	Incremental wetted area of airplane ratioed to fuselage wetted area
$(T/A)_{NET}$	Net tail rotor disc loading (psf)
TAS	True airspeed (knots)
T_{AUX}/T_{TOT}	Ratio of thrust produced by auxiliary propulsive device to total helicopter thrust required
TFEF	Tail fin aspect ratio effectiveness factor
TFI	Turbine inlet temperature (flight idle power setting), or input on engine cycle sheets
TGI	Turbine inlet temperature (ground idle power setting), or input on engine cycle sheets

TMAX	Turbine inlet temperature (maximum power setting), or input on engine cycle sheets
TMIL	Turbine inlet temperature (military power setting), or input on engine cycle sheets
TNP	Turbine inlet temperature (normal power setting) or input on engine cycle sheets
T/W	Configuration thrust/weight ratio (hover)
$(T/W)_D$	Configuration design thrust/weight ratio (hover)
$(T_{AUX}/T_{TOT})_C$	Ratio of thrust produced by auxiliary propulsive device to total helicopter thrust required. This value input as design point for sizing primary or auxiliary independent engines in cruise.
ΔT_{inCE}	Increment in ambient temperature for primary engine sizing at cruise condition ($^{\circ}R$)
ΔT_{inTO}	Increment in ambient temperature for engine sizing at takeoff conditions ($^{\circ}R$)
T/ θ	Referred turbine temperature ($^{\circ}R$)
t_O	Initial time at start of mission (hours)
t_T	Incremental time for taxi (hours)
t_H	Incremental time for hover (hours)
t_L	Incremental time for loiter (hours)
t_{FW}	Incremental time for change of fuel weight (hours)
t_{PW}	Incremental time for change of payload weight (hours)
Δt_H	Step size for hover (hours)
Δt_L	Step size for loiter (hours)
$(t/C)_R$	Wing root thickness to chord ratio
$(t/C)_T$	Wing tip thickness to chord ratio
$(t/C)_{HT}$	Horizontal tail mean thickness to chord ratio

$(t/C)_{VT}$	Vertical tail mean thickness to chord ratio
$(t/C)_{R_A}$	Aft rotor pylon root thickness to chord ratio
$(t/C)_{R_F}$	Forward rotor pylon root thickness to chord ratio
$(t/C)_{T_A}$	Aft rotor pylon tip thickness to chord ratio
$(t/C)_{T_F}$	Forward rotor pylon tip thickness to chord ratio
$(t/C)_{.25R}$	Main rotor blade thickness to chord ratio @0.25 rotor radius
V_C	Design cruise speed for engine sizing (kts)
V_{CEH1}	Main rotor vertical R/C efficiency factors
V_{CEH2}	
V_{DES}	Flight speed at which single rotor helicopter vertical tail is sized to provide complete directional stability in the event of the loss of the tail rotor (kt)
V_{DIVE}	Dive speed (knots EAS)
V_{in}	True airspeed for cruise during cruise segment with CRSIND = 2 (kt)
V_{MO}	Maximum operating equivalent airspeed (kt)
$(V_{R/C})_D$	Design vertical rate of climb (ft/min) used in sizing primary engines in hover
$V_{R/C}$	Vertical rate of climb (ft/min)
V_{TIP}	Main rotor design tip speed (pfs). Note: This is the tip speed corresponding to $N_{II} = N^*_{II}$
V_{TTR}	Tail rotor design tip speed (hover) -(fps)
V_{TIP_P}	Propeller design tip speed (fps)
$\bar{V}_H$	Horizontal tail volume coefficient
W_{APU}	Auxiliary power unit weight (lb)

W_{AV}	Avionics weight (lb)
W_{FE}	Weight of fixed equipment (lb)
W_{FUL}	Weight of fixed useful load (lb)
W_{FURN}	Furnishings and equipment weight (lb)
W_{G_O}	Initial gross weight at start of mission (lb)
W_i	Weight of inboard store
W_o	Weight of outboard store
W_{PL}	Weight of payload (lb)
W/A	Disc loading(psf)
W/S	Wing loading (psf)
$\dot{W}_{MAX}/\dot{W}^*$	Fuel flow limit - ratio of maximum fuel flow to fuel flow at maximum static power, sea level, standard
ΔW_{FC}	Flight controls group incremental weights (lb)
ΔW_P	Propulsion group incremental weight (lb)
ΔW_{ST}	Structures group incremental weight (lb)
ΔW_f	Increment in fuel weight during change of fuel weight subroutine (lb)
ΔW_{PL}	Increment in payload weight during change of payload weight subroutine (lb)
δW_f	Fuel required additive reserve factor
W_F	Width of fuselage (ft)
$\dot{W}_f/\delta\sqrt{\theta} F_N^*$	Referred fuel flow for turbojet/fan engine cycles
$\dot{W}_f/\delta\sqrt{\theta} SHP^*$	Referred fuel flow rate for primary engines (turboshaft engine cycles, lb/hr/SHP*)

$X_{ADS}/\Delta X_{ADS}$

Auxiliary independent drive system C.G. position aft of nose as a fraction of the distance between the auxiliary independent engine and the propeller

X_{AE}/l_B

Auxiliary independent engine C.G. position aft of nose as a fraction of body length

X_{APU}/l_B

Auxiliary power unit C.G. position aft of nose as a fraction of body length

X_{AR}

Propeller blade attachment point as a fraction of propeller radius

X_{AR}/l_{TB}

Propeller position aft of body/tail boom junction as a fraction of tail boom length

X_{ASC}/l_B

Auxiliary rotor (propeller) systems controls C.G. position aft of nose as a fraction of body length

X_{AV}/l_B

Avionics C.G. position aft of nose as a fraction of body length

X_{CGF}/l_B

Single rotor fuselage (minus tail boom) C.G. position aft of nose as a fraction of body length

X_{CMR}

Main rotor blade cutout (end of blade shank, beginning of rotor airfoil sections) position as a fraction of rotor radius

X_{CTR}

Tail rotor blade cutout (end of blade shank, beginning of rotor airfoil sections) position as a fraction of rotor radius

X_{FURN}/l_B

Furnishings and equipment C.G. position aft of nose as a fraction of body length

X_M/l_B

Ratio of distance from tip of nose to rotor shaft, X_M , to main fuselage length (B) (single rotor helicopter)

X_{MG}/l_B

Main landing gear position aft of nose as a fraction of body length

X_{MR}

Main rotor blade attachment point as a fraction of rotor radius

X_{NG}/l_B

Nose landing gear position aft of nose as a fraction of body length

X_{PE}/ℓ_B	Primary engine C.G. position aft of nose as a fraction of body length
$X_{PDS}/\Delta X_{PDS}$	Primary drive system C.G. position aft of nose as a fraction of the distance between main and tail rotor centers
X_{SC}/ℓ_B	Rotor system controls C.G. position aft of nose as a fraction of body length
X_{TR}	Tail rotor blade attachment point as a fraction of rotor radius
X_1/ℓ_P	Distance of forward rotor center from aircraft nose as a fraction of aircraft nose section length
X_2/ℓ_T	Distance of aft rotor center from aircraft tail cone as a fraction of aircraft tail section length
Y_{CL}	Clearance from inboard propeller tip to inboard propeller tip across fuselage (ft)
$\left. \begin{matrix} z_1 \\ z_2 \\ z_3 \end{matrix} \right\}$	Primary engine nacelle dimensional factors
z_4	
$\left. \begin{matrix} z_5 \\ z_6 \end{matrix} \right\}$	
z_5	Auxiliary independent engine nacelle dimensional factors
z_6	
α_{LO}	Wing airfoil section angle of zero lift
$\Lambda C/4$	Sweep angle of wing quarter chord (degrees)
λ	Taper ratio of wing
λ_{AP}	Taper ratio of aft rotor pylon
λ_{FP}	Taper ratio of forward rotor pylon
λ_H	Taper ratio of horizontal tail
λ_{VT}	Taper ratio of vertical tail

η_{P3}	Propeller propulsive efficiency for SGTIND = 3
η_{P5}	Propeller propulsive efficiency for SGTIND = 5
η_{P4}	Propeller propulsive efficiency for SGTIND = 4, 6 tabular function of Mach number
η_T	Transmission efficiency
$\eta_{T\text{AUX}}$	Transmission efficiency (auxiliary drive system)
ζ_{VT}	Vertical tail span overlap distance/tail rotor radius ratio - input as a function of tail rotor radius
ζ_2	Propeller over wing tip overlap (fraction of radius)
θ	Ambient temperature ratio, tabular function of altitude
θ_{TMR}	Main rotor blade twist (degrees)
θ_{TR}	Tail rotor blade twist (degrees)
ξ_4	Primary engine dimensional factor
$\dot{\psi}$	Helicopter yaw rate, rad/sec
$\ddot{\psi}$	Helicopter yaw acceleration, rad/sec ²
σ_{MR}	Main rotor solidity ($\sigma = bc/\pi R$)
σ_{TR}	Tail rotor solidity ($\sigma = bc/\pi R$)
μ	Rotor forward flight advance ratio ($\mu = V_{FPS}/V_{TIP}$)

5.3.2 Program Indicators

Option Indicators

OPTIND	1 = size aircraft
	2 = calculate performance (specify initial gross weight)
	3 = calculate performance (specify operating weight empty)
OPTIONAL PRINT	0 = standard print
	1 = detailed print

Propulsion Indicator

AIPIND	1 = single gas generator connected to main rotor and auxiliary propulsion
	2 = independent gas generators
ENGIND	0 = turboshaft (power producing) cycle
	1 = turbofan or turbojet (thrust producing) cycle
ESCIND	1 = program will size engines for takeoff only
	2 = program will size engines for more critical choice of takeoff or cruise

NOTE: ESCIND is applicable only if FIXIND = 1

FIXIND	0 = fixed size engines, user inputs maximum power
	1 = rubberized engines, program will calculate maximum power
FIXINDI	0 = user inputs fixed size auxiliary independent engines
	1 = program sizes auxiliary independent engines to meet cruise requirement
NIIND, NIINDI	0 = no N_I limit
	1 = N_I limit

N1 θ IND,
 N1 θ INDI 0 = no $N_I/\sqrt{\theta_1}$ limit
 1 = $N_I/\sqrt{\theta_1}$ limit

N2IND,
 N2INDI 0 = no N_{II} limit, engine operating at optimum N_{II}
 1 = N_{II} limit, engine operating at known value of N_{II} (in general nonoptimum)

POWIND 0 = maximum engine rating
 1 = military engine rating

QIND,
 QINDI 2 = normal engine rating
 0 = no torque limit
 1 = torque limit

RNOIND,
 RNOINDI 0 = no Reynolds number corrections
 1 = Reynolds number corrections

WDTIND,
 WDTINDI 0 = Reynolds number corrections
 1 = fuel flow limit

ROTIND 1 = performance calculated by short method
 2 = rotor map input, corrections applied
 3 = rotor map input, no corrections applied

npIND 0 = user specifies "point" propeller efficiencies for climb, cruise and descent
 1 = user inputs propeller performance map
 2 = propeller performance automatically calculated within program

Aerodynamics Indicators

DRGIND 1 = drag build up by component, Reynolds number scaling
 2 = drag trend by input GW/F_e versus GW

OSWIND 0 = user inputs Oswald's efficiency (e)
 1 = program calculates Oswald's efficiency

Sizing Indicators

APHIND	1 = input aft pylon height 2 = input gap/stagger ratio
AUXIND	1 = pure helicopter 2 = including wings (only) 3 = including auxiliary propulsion (only) 4 = compound (wings and auxiliary propulsion)
b _W IND	1 = input span/diameter ratio 2 = input wing aspect ratio 3 = determined for propeller clearance
CNFIND	1 = single rotor 2 = tandem rotor
FDMIND	1 = input overlap and rotor positions 2 = input overlap and cabin length 3 = input cabin length and rotor positions
HTIND	0 = no horizontal tail 1 = input fixed size tail 2 = input tail volume coefficient
MRPIND	0 = user inputs main rotor placement (single rotor) 1 = program calculates main rotor positions based on simple mass balance 2 = same as 1, except that in the case of a compound helicopter, the auxiliary drive system, propeller and auxiliary independent engines (if any) are assumed to be located on the wing.
RDMIND	1 = input main rotor diameter, σ

- 2 = input W/A , σ
- 3 = input diameter, C_T/σ
- 4 = input W/A , C_T/σ
- SWIND**
- 1 = input wing area
- 2 = input wing loading
- 3 = size for maneuver
- TRDIND**
- 1 = use trend of diameter main/diameter
tail = $f_n(W/A)_{MAIN}$
- 2 = input diameter
- 3 = input T/A
- TRSIND**
- 1 = input σ
- 2 = input C_T/σ
- VTFIND**
- 1 = input aspect ratio and tail fin overlap
- 2 = input directional stability required
and tail fin overlap
- 3 = input directional stability required
and aspect ratio
- XMSNIND**
- 1 = main, tail and auxiliary drive system
ratings specified as fraction of primary
engine installed power (in the case of a
compound helicopter with auxiliary inde-
pendent propulsion, the auxiliary
independent drive system rating is
specified as a fraction of the auxiliary
independent engine installed power)
- 2 = main, tail, and auxiliary drive system
ratings specified at fraction of power
required to hover or cruise at design
conditions (more critical of the two
conditions is selected)
- 3 = same as 2, except the most critical of
the two design conditions is compared to
the drive system rating required at an
alternate payload/gross weight hover at
the design point conditions, the most
critical of these three conditions is
selected

Flight Path Control Indicators

h_{OPT}IND 0 = cruise segments performed at specified altitude

 1 = cruise segments preceded by climb or transfer altitude are performed at optimum altitude, constrained by an input maximum altitude

Mission Performance Indicators

CLMIND 1 = climb at maximum R/C

 2 = climb at constant EAS

 3 = climb at constant Mach No.

 4 = climb at constant TAS

CRSIND 1 = cruise at cruise power

 2 = cruise at constant true airspeed

 3 = cruise at speed for best specific range

 4 = cruise at speed for 99% of best specific range

 5 = cruise - climb (constant W/δ) at speed for best specific range

 6 = cruise - climb (constant W/δ) at speed for 99% of best specific range

DESIND 1 = descend at constant TAS

 2 = descend at constant EAS

 3 = descend at constant Mach No.

RMAXND 0 = descent flight path ends at specified terminal range (cruise segment must be input previous to descent)

 1 = checks specified terminal range, if predicted flight path will end beyond specified terminal range value, spiral descent path assumed at that point, if predicted flight path ends before reaching specified terminal range point, program prints SHALLOWER DESCENT REQUIRED

2 = descent ends at specified minimum altitude, terminal range requirement not considered

3 = fuel used and time required for descent calculated but no range credit given (i.e., spiral descent path)

SGTIND

0 = end of mission

1 = taxi

2 = takeoff, hover and landing

3 = climb

4 = cruise

5 = descent

6 = loiter

7 = change of fuel weight

8 = change of payload weight

9 = transfer altitude

100 = end of case

NOTE: Segments 1 through 6 can be used for reserve fuel calculations (gross weight reset following segment) by inputting 10X SGTIND; i.e., SGTIND = 10, 20, 30, 40, 50, or 60

TOLIND

1 = user inputs required thrust-to-weight ratio and $V_{R/C}$

2 = user inputs required fractions of maximum power and $V_{R/C}$

3 = same as 1 but analysis includes hover-in-ground effect

4 = same as 2 but analysis includes hover-in-ground effect

WGTIND

0 = restriction on maximum aircraft weight, weight cannot exceed gross weight.

1 = no restriction on aircraft weight (will only apply when running performance)

Atmosphere Indicator

ATMIND 0 = standard atmosphere

 1 = non-standard atmosphere, user inputs
 single point value for increment in
 ambient temperature above standard day
 value

 2 = non-standard atmosphere, user inputs
 table of temperature ratio as a function
 of altitude

6.0 PROGRAM OUTPUT

A reproduction of the program output for two sample cases is included in Section 7.0. The following discussion describes the program printout in general and lists the diagnostic error printouts which are possible.

6.1 DESCRIPTION OF PRINTOUT

The printout for HESCOMP consists of four types of information:

- a. General
- b. Input Data
- c. Sizing Data (program output)
- d. Mission Performance Data (for the "sized" helicopter)

The general information (item a) is printed out at the beginning of each new case. Each of the other groupings (input, sizing data, and performance data) starts on a new page. For cases with OPTIND = 2 or 3 (performance only), the sizing data is not printed out. The printout is described in detail below.

6.1.1 General Printout

6.1.1.1 Fixed Heading:

HESCOMP

HELICOPTER SIZING AND PERFORMANCE COMPUTER PROGRAM

Depending on the configuration options (CNFIND, AUXIND, AIPIND, ENGIND) chosen, one of the following statements will be printed out.

SINGLE ROTOR PURE HELICOPTER

SINGLE ROTOR WINGED HELICOPTER

SINGLE ROTOR COMPOUND HELICOPTER

SINGLE ROTOR COMPOUND HELICOPTER AUXILIARY INDEPENDENT
T/SHAFT CRUISE PROPULSION

SINGLE ROTOR COMPOUND HELICOPTER AUXILIARY INDEPENDENT T/FAN
OR T/JET CRUISE PROPULSION

SINGLE ROTOR AUXILIARY PROPULSION HELICOPTER

SINGLE ROTOR AUXILIARY PROPULSION HELICOPTER AUXILIARY
INDEPENDENT T/SHAFT CRUISE PROPULSION

SINGLE ROTOR AUXILIARY PROPULSION HELICOPTER AUXILIARY
INDEPENDENT T/FAN OR T/JET CRUISE PROPULSION

The printout for the tandem rotor configurations will be identical except for the substitution of TANDEM ROTOR for SINGLE ROTOR

6.1.1.2 Arbitrary Heading

An arbitrary heading may be input by the user on a title card (see Section 5.2, input sheet for general information).

6.1.2 Input Data

All program input data is printed out as it appears on the data cards. Seven columns are printed. These correspond to the first location on the card, the number of variables on the card (from 1 to 5), and the values of these variables. With this information and a copy of the input sheets it is possible to determine the input value for any variable.

6.1.3 Sizing Data

This group is printed out only if OPTIND = 1. The data is represented by a symbol, followed by a written description, followed by the value with the units. For example:

WG/A	DISC LOADING	10.0 LB/SQFT
------	--------------	--------------

The data is printed out in groups, each group having a heading. The specific variables which are printed out will depend upon certain options chosen. Notations are made in the following list to show where this occurs.

6.1.3.1 Dimensional Data

Single Rotor Helicopter

Fuselage:

$l_F, l_C, l_B, l_{TB}, X_M, w_F, S_F$

Wing:

If AUXIND = 2 or 4 print AR, $S_W, b_W, C_W, \lambda_{C/4}, \lambda, (t/c)_T,$

$w_G/S_W, G_{RW}, C_F/C$

If AUXIND = 1 or 3 print NO WING USED

Horizontal Tail:

If HTIND = 1 or 2 print AR_{HT} , S_{HT} , b_{HT} , C_{HT} , λ_{HT} , L_{TH}

If HTIND = 0 print NO HORIZONTAL TAIL USED

Vertical Tail:

AR_{VT} , S_{VT} , b_{VT} , C_{VT} , λ_{VT} , Z_{TR} , ζ_{VT} , $(T/C)_{VT}$

Main Rotor Pylon:

AR , S_{FP} , FA_{FP} , H_{P1} , C_{FP} , λ_{FP} , $(T/C)_R$, $(T/C)_T$

Primary Engine Nacelle:

ℓ_N , D_N , S_N

Auxiliary Independent Engine Nacelle:

If AIPIND = 2 print L_{NI} , D_{NI} , S_{NI}

Auxiliary Independent Engine Nacelle Strut:

If AIPIND = 2 print S_{STR} , b_{NS} , C_{NS}

Auxiliary Independent Propulsion:

If AIPIND = 2 and ENGIND = 1, print AUXILIARY INDEPENDENT PROPULSION - TURBOFAN (OR TURBOJET) ENGINE

If AIPIND = 1 print NO AUXILIARY INDEPENDENT ENGINE USED

Propeller (Auxiliary Propulsion):

If AUXIND = 3 or 4 and ENGIND = 0 print D_{AR} , AF , σ_{AR} , N_{RA} , NO. BLADES, V_{TIP}

If AUXIND = 1 or 2 print NO PROPELLER USED

Main Rotor:

D_{MR} , σ_{MR} , W_G/A , C_T/σ , N_R , NO BLADES, θ_{MR} , X_C , V_{TIP}

Tail Rotor:

D_{TR} , σ_{TR} , $(T/A)_{NET}$, C_T/σ , NO BLADES, θ_{TR} , X_{CTR} , G , V_{TIP}

Tandem Rotor Helicopter

Fuselage:

$\ell_F, \ell_C, \Delta X_1, \Delta X_2, w_F, G/S, (O/L/D), S_F$

Wing:

Same printout as single rotor helicopter

Forward Rotor Pylon:

$AR, S_{FP}, FA_{FP}, H_{P1}, C_{FP}, \lambda_{FP}, (T/C)_R, (T/C)_T$

Aft Rotor Pylon:

$AR, S_{AP}, H_{P2}, C_{AP}, \lambda_{AP}, (T/C)_R, (T/C)_T$

Primary Engine Nacelle:

ℓ_N, D_N, S_N

Auxiliary Independent Engine Nacelle:

Same printout as single rotor helicopter

Auxiliary Independent Engine Nacelle Strut:

Same printout as single rotor helicopter

Auxiliary Independent Propulsion:

Same printout as single rotor helicopter

Propeller (Auxiliary Propulsion):

Same printout as single rotor helicopter

Main Rotor:

Same printout as single rotor helicopter

6.1.3.2 Weights Data

Single and Tandem Rotor Helicopters

First print M_{LF}, G_{LF}, U_{LF} , then print,

Propulsion Group:

W_{PRG} , $K_{12}W_{PRB}$, $K_{13}W_{PH}$, W_{BF} , $K_{15}W_{AR}$, W_{DS} , $K_{16}W_{PDS}$,
 $K_{20}W_{TRDS}$, $K_{17}W_{ADS}$, $K_{18}W_{EP}$, $K_{19}W_{EA}$, W_{PEI} , W_{AEI} , W_{FS} , ΔW_P ,
 W_P

Structures Group:

K_8W_W , W_{TG} , K_9W_{HT} , $K_{14}W_{TR}$, K_6W_B , K_7W_{LG} , W_{NG} , W_{MG} , W_{TES} ,
 W_{PES} , W_{AES} , ΔW_{ST} , W_{ST}

Flight Controls Group:

W_{PFC} , W_{CC} , K_1W_{RC} , K_2W_{SC} , K_3W_{FW} , W_{TM} , W_{SAS} , W_{AFC} , K_4W_{RCA} ,
 K_5W_{SCA} , W_{MC} , ΔW_{FC} , W_{FC}

Weight of Fixed Equipment:

W_{FE}

Weight Empty

W_E

Fixed Useful Load

W_{FUL}

Operating Weight Empty

OWE

Payload

W_{PL}

Fuel

$(W_F)_A$

Gross Weight

WG

6.1.3.3 Rotor Data

Single Rotor Helicopter

ROTOR CYCLE NO. _____
(printed if ROTIND = 1)

ROTOR MAP NO. _____
(printed if ROTIND = 2)

FIXED MAIN ROTOR SOLIDITY INPUT
(printed if RDMIND = 1 or 2)

If RDMIND = 3 or 4, and depending on which solidity sizing requirement is most critical, one of the following statements will be printed out:

MAIN ROTOR SOLIDITY SIZED BY MANEUVER CONDITIONS
H = _____ FT., TEMP. = _____ DEG., V = _____ KT.
ROTOR MANEUVER G'S = _____, C_T/σ = _____

MAIN ROTOR SOLIDITY SIZED BY HOVER CONDITIONS
H = _____ FT., TEMP. = _____ DEG., T/W = _____
 C_T/σ = _____

MAIN ROTOR SOLIDITY SIZED BY CRUISE CONDITIONS
H = _____ FT., TEMP. = _____ DEG., V = _____ KT.
ROTOR LIFT/GW FRACTION = _____ C_T/σ = _____

Which is followed by:

FIXED TAIL ROTOR SOLIDITY
(printed if TRSIND = 1)

TAIL ROTOR SIZED AT _____ TIMES THE SOLIDITY REQUIRED TO
SATISFY HOVER ANTI-TORQUE REQUIREMENTS AT
H = _____ FT., TEMP. = _____ DEG.F., C_{TG}/C_{TNET} = _____
(printed if TRSIND = 2 and $\psi = 0$, $\ddot{\psi} = 0$)

TAIL ROTOR SIZED AT _____ TIMES THE SOLIDITY REQUIRED TO
SATISFY HOVERING TURN REQUIREMENTS AT

H	=	_____	FT.
TEMP	=	_____	F.
C_{TG}/C_{TNET}	=	_____	DEG.
YAW RATE	=	_____	RAD/SEC
YAW ACCELERATION	=	_____	RAD/SEC ²
TAIL ROTOR POLAR MOM. OF INERTIA	=	_____	SLUG/FT ²
HELICOPTER YAW MOM. OF INERTIA	=	_____	SLUG/FT ²

(printed if TRSIND = 2 and $\dot{\psi}$ or $\ddot{\psi} \neq 0$.)

Tandem Rotor Helicopter

Main rotor data printout same as for single rotor helicopter.

6.1.3.4 Propulsion Data

Single Rotor Helicopter

PRIMARY PROPULSION CYCLE NO. _____
TURBOSHAFT ENGINE

_____ ENGINES

BHP*P MAX. STANDARD S.L. STATIC H.P. = _____ H.P.

ENGINE SIZE WAS FIXED BY INPUT
(printed if FIXIND = 0)

IF ESCIND = 1 print;

ENGINE SIZED FOR TAKEOFF AT T/W = _____,
H = _____ FT., TEMP. = _____ DEG.F.
AND _____ ENGINES INOPERATIVE

If ESCIND = 2 either of the following statements are printed depending on which engine sizing requirement (hover or cruise) is critical.

ENGINE SIZED FOR TAKEOFF AT T/W = _____,
H = _____ FT., TEMP. = _____ DEG.F.
AND _____ ENGINES INOPERATIVE

ENGINE SIZED FOR CRUISE AT V_C = _____ KNOTS
H = _____ FT., TEMP. = _____ DEG.F.
AND _____ ENGINES INOPERATIVE

Which is followed by:

NO AUX. INDEPENDENT ENGINE CYCLE SELECTED
(printed if AUXIND = 3 or 4 and AIPIND = 1)

AUX. INDEPENDENT PROPULSION CYCLE NO. _____
(printed if AUXIND = 3 or 4 and AIPIND = 2)

IF ENGIND = 0.0, TURBOSHAFT ENGINE is printed
IF ENGIND = 1.0, either TURBOFAN or TURBOJET ENGINE is printed

_____ ENGINES

BHP*P MAX. STANDARD S.L. STATIC H.P. _____ H.P.
(printed if ENGIND = 0.0)

T*P MAX. STANDARD S.L. STATIC THRUST _____ LBS.
(printed if ENGIND = 1.0)

ENGINE SIZE WAS FIXED BY INPUT
(printed if FIXINDI = 0.0)

ENGINE SIZED FOR CRUISE AT V_C _____ KNOTS
H = _____ FT., TEMP. = _____ DEG.F.
(printed if FIXINDI = 1.0)

MAIN DRIVE SYSTEM RATING _____ H.P.
MAIN ROTOR DRIVE SYSTEM RATING _____ H.P.

Depending on the transmission sizing option chosen (XMSNIND),
and the results of the sizing, one of the following four
statements will be printed:

XMSN SIZED AT _____ PERCENT OF TOTAL PRIMARY
ENGINE INSTALLED POWER
MAX. STANDARD S.L. STATIC H.P.
(printed if XMSNIND = 1.0)

XMSN SIZED AT _____ PERCENT OF MAIN ROTOR HOVER
POWER REQUIRED AT
H = _____ FT., TEMP. = _____ DEG.F.
(printed if XMSNIND = 2 or if XMSNIND = 3 and the
alternate payload hover is not critical)

XMSN SIZED AT _____ PERCENT OF MAIN ROTOR HOVER
POWER REQUIRED AT ALTERNATE
PAYLOAD = _____ LBS., ALTERNATE GROSS WEIGHT = _____ LBS.
H = _____ FT., TEMP. = _____ DEG.F.
(printed if XMSNIND = 3 and the alternate payload hover
is critical)

XMSN SIZED AT _____ PERCENT OF MAIN ROTOR CRUISE
POWER REQUIRED AT V_C = _____ KT.,
H = _____ FT., TEMP. = _____ DEG.F.
(printed if XMSNIND = 2 or if XMSNIND = 3 and the
alternate payload hover is not critical)

Which is followed by:

TAIL ROTOR DRIVE SYSTEM RATING _____ H.P.
(printed if CNFIND = 1)

Depending on the transmission sizing option chosen (XMSNIND)
and the results of the sizing, one of the following four
statements will be printed:

XMSN SIZED AT _____ PERCENT OF TOTAL PRIMARY ENGINE
INSTALLED POWER
MAX. STANDARD S.L. STATIC H.P.
(printed if XMSNIND = 1.0)

XMSN SIZED AT _____ PERCENT OF TAIL ROTOR HOVER
POWER REQUIRED
AT H = _____ FT., TEMP. = _____ DEG.F.
(printed if XMSNIND = 2 or if XMSNIND = 3 and the
alternate payload hover is not critical)

XMSN SIZED AT _____ PERCENT OF TAIL ROTOR HOVER
POWER REQUIRED AT ALTERNATE
PAYLOAD = _____ LBS., ALTERNATE GROSS WEIGHT = _____ LBS.
H = _____ FT., TEMP. = _____ DEG.F.
(printed if XMSNIND = 3 and the alternate payload
hover is critical)

XMSN SIZED AT _____ PERCENT OF TAIL ROTOR CRUISE
POWER REQUIRED AT V_C = _____ KT.
H = _____ FT., TEMP. = _____ DEG.F.
(printed if XMSNIND = 2 or if XMSNIND = 3 and the
alternate payload hover is not critical)

Which is followed by:

AUXILIARY PROPULSION DRIVE SYSTEM RATING _____ H.P.
(printed if AUXIND = 3 or 4 and AIPIND = 1)

Depending on the transmission sizing option chosen (XMSNIND)
and the results of the sizing one of the following three
statements will be printed:

XMSN SIZED AT _____ PERCENT OF TOTAL CONFIGURATION
POWER REQUIRED TO HOVER
AT H = _____ FT., TEMP. = _____ DEG.F.
(printed if ESCIND = 1.0 and XMSNIND = 2 or 3)

XMSN SIZED AT _____ PERCENT OF TOTAL PRIMARY ENGINE
INSTALLED POWER
MAX. STANDARD S.L. STATIC H.P.
(printed if XMSNIND = 1.0)

XMSN SIZED AT _____ PERCENT OF AUX. PROPULSION
CRUISE POWER REQUIRED AT V_C = _____ KT.
 H_C = _____ FT., TEMP. = _____ DEG.F.
(printed if ESCIND = 2.0 and XMSNIND = 2 or 3)

AUXILIARY INDEPENDENT PROPULSION DRIVE SYSTEM RATING
(printed if AUXIND = 3 or 4, AIPIND = 2 ENGIND = 0)

Depending on the transmission sizing option chosen (XMSNIND) and the results of the sizing one of the following four statements will be printed:

XMSN SIZED AT _____ PERCENT OF TOTAL AUXILIARY
INDEPENDENT ENGINE INSTALLED POWER
MAX. STANDARD S.L. STATIC H.P.
(printed if AIPIND = 2, ENGIND = 0, and XMSNIND = 1.0)

XMSN SIZED AT _____ PERCENT OF MAX. AUXILIARY
INDEPENDENT ENGINE POWER AVAILABLE
AT H = _____ FT., TEMP. = _____ DEG.F.
(printed if AIPIND = 2, ENGIND = 0, ESCIND = 1 and
XMSNIND = 2 or 3)

XMSN SIZED AT _____ PERCENT OF MAX. AUXILIARY
INDEPENDENT ENGINE POWER AVAILABLE IN CRUISE
AT V_C = _____ KT., H_C = _____ FT., TEMP. = _____ DEG.F.
(printed if AIPIND = 2, ENGIND = 0, XMSNIND = 2 or 3
and FIXIND = 0.0)

XMSN SIZED AT _____ PERCENT OF AUXILIARY PROPULSION
CRUISE POWER REQUIRED AT V_C = _____ KT.
 H_C = _____ FT., TEMP. = _____ DEG.F.
(printed if AIPIND = 2, ENGIND = 0, XMSNIND = 2 or 3
and FIXIND = 1.0)

6.1.3.5 Aerodynamics Data

Single and Tandem Rotor Helicopter

TOTAL EFFECTIVE FLAT PLATE AREA
TOTAL WETTED AREA
MEAN SKIN FRICTION COEFF.

DRAG BREAKDOWN

WING FE
FUSELAGE FE
FORWARD (MAIN) ROTOR PYLON FE
AFT ROTOR PYLON FE
MAIN ROTOR HUB(s) FE
TAIL ROTOR HUB FE
VERTICAL TAIL FE
HORIZONTAL TAIL FE
PRIMARY ENGINE NACELLE FE
AUXILIARY INDEPENDENT CRUISE ENGINE NACELLE FE
AUXILIARY INDEPENDENT CRUISE ENGINE NACELLE STRUT FE
INCREMENTAL FE

AERODYNAMIC COEFFICIENTS

A5

A6

A7

A8

A9

WING LIFT EFFICIENCY FACTOR

VERTICAL TAIL LIFT EFFICIENCY

6.1.4 Mission Performance Data

Two types of output are possible. If the OPTIONAL PRINT INDICATOR = 0, a standard printout will occur. If the indicator is input as 1, a detailed printout will occur. This will include all data printed in the standard printout plus additional information.

6.1.4.1 Standard Printout

The mission performance data is printed out by segment in chronological sequence. Up to 15 columns of data are printed out depending upon the segment. For all segments, the following information is printed:

t: time in hours

R: range in nautical miles

W_f : weight of fuel used in pounds

W: aircraft weight in pounds

h: altitude in feet

TAS: the true airspeed in knots

Primary Turb. Temp: the primary engine turbine temperature

PRIMARY ENGINE CODE: a code letter which designates the condition governing the engine performance:

P = power (or thrust) required

T = turbine temperature
(engine rating)

W = fuel flow limit

N1 = gas generator shaft rpm limit

C = compressor (N_I/θ_1) limit

N2 = output shaft rmp limit

Q = torque limit

PRIMARY ENG. PEHF: The primary engine horsepower fraction. This is the ratio of power being used at any altitude, Mach number condition to the maximum power available at that condition.

In addition, the following data is printed out in different segments:

AUX. TURB. TEMP: The auxiliary independent engine turbine temperature.

AUX. ENG. CODE: A code letter which designates the condition governing the auxiliary independent engine performance: (code is same as for primary engines).

AUX. ENG. PEHF: The auxiliary independent engine thrust or horsepower fraction. This is the ratio of thrust or power being used at any altitude, Mach number condition to the maximum thrust, or power available at that condition.

AUX. ENG. FUEL FLOW: Auxiliary independent engine time rate of fuel consumption in pounds per hour.

TOTAL FUEL FLOW: Total time rate of fuel consumption (primary plus auxiliary independent engines) in pounds per hour.

T/W: The thrust-to-weight ratio (printed out in takeoff, hover, and landing).

FM: Main rotor overall hover figure of merit (for a tandem rotor configuration, this includes rotor/rotor interference) (printed out in takeoff, hover and landing).

BHP: Total power required (printed out in takeoff, hover, and landing, climb, cruise, descent, and loiter).

CT: Main rotor thrust coefficient (printed out in takeoff, hover, and landing, climb, cruise, descent, and loiter).

CT/SIGMA: CT/main rotor solidity (printed out in takeoff, hover, and landing).

EAS: The equivalent airspeed in knots (printed out in climb, cruise, descent, and loiter).

MU: Main rotor advance ratio (printed out in climb, cruise, descent, and loiter).

CT PRIME/SIGMA: Main rotor cruise lift coefficient/main rotor solidity (printed out in climb, cruise, descent, and loiter).

ALPHA D/L: Angle of total rotor thrust (lift plus propulsive force) vector with respect to a line perpendicular to the A/C flight path (printed out in climb, cruise, descent and loiter).

NMPP: The specific range in nautical miles per pound (printed out in cruise).

GAMMA: The flight path angle in degrees (printed out in climb and descent).

R/C: Rate of climb in feet per minute (printed out in climb).

R/S: Rate of descent in feet per minute (printed out in descent).

6.1.4.2 Detailed Printout

In addition to the data printed above, the following data (unless noted otherwise) will be printed in takeoff, hover, and landing, climb, cruise, descent, and loiter segments if the OPTIONAL PRINT INDICATOR = 1:

VRC RHP: Vertical rate of climb rotor horsepower (printed out only in takeoff, hover and landing).

FMI: Isolated main rotor hover figure of merit (printed out only in takeoff, hover, and landing).

TOTAL FUEL FLOW: Total fuel consumption (primary + auxiliary independent engines) - lb/hr (printed out only in loiter).

M. ROTOR VTIP: Main rotor tip speed - feet per second

M. ROTOR RHP: Main rotor horsepower (no losses)

T. ROTOR VTIP: Tail rotor tip speed - feet per second

T. ROTOR RHP: Tail rotor horsepower (no losses)

PRIM. ENG. FUEL FLOW: Primary engine fuel consumption - lb/hr

AUX. ENG. FUEL FLOW: Auxiliary independent engine fuel consumption - lb/hr

ROTLIM CODE: A code letter which designates whether main rotor has exceeded the rotor limits input to the program.

A = Within input rotor limits

E = Rotor limits exceeded

DELCDM: Compressibility drag coefficient increment to rotor profile power. In hover, it is a function of rotor C_T/σ and VTIP. In cruise it is a function of rotor C_T/σ and advancing blade tip Mach number (only printed out when a rotor "cycle" is input).

CPPRO: Rotor profile power coefficient (only printed out when a rotor "cycle" is input).

CPIND: Rotor induced power coefficient (only printed out when a rotor "cycle" is input).

CDO: Rotor profile drag (total) coefficient (only printed out when a rotor "cycle" is input).

PROP VTIP: Propeller tip speed - ft/sec.

BHP AUX: Auxiliary propulsion power required (not printed out in takeoff, hover, and landing).

ETAP PROP: Propeller cruise efficiency

TAUX/T:	Ratio of auxiliary propulsion thrust to total configuration thrust required.
AUX. ENG. FUEL FLOW:	} Same as noted earlier
AUX. TURB. TEMP:	
AUX. ENG. CODE:	
AUX. ENG. PEHF:	
AUX. ENG. BHP OR THRUST:	Auxiliary independent engine power (if ENGIND = 0, horsepower required printed out. If ENGIND = 1, thrust required printed out).
CPPAR:	Rotor parasite power coefficient (only printed out when a rotor "cycle" is input).
CPNUD:	Rotor nonuniform downwash power coefficient (only printed out when a rotor "cycle" is input).
DELCDS:	Retreating blade stall coefficient increment to rotor profile power (only printed out when a rotor "cycle" is input).
CXR:	Rotor propulsive force coefficient
J	Propeller advance ratio
CP	Propeller power coefficient
CT	Propeller thrust coefficient
CLW	Wing lift coefficient
CDW	Wing profile drag coefficient
RN	Fraction of total lift carried by rotor

6.1.4.3 Headings

At the beginning of each segment, a printout will identify the segment data which follows. The following messages can be printed:

a. TAXI FOR _____ HRS. AT GROUND IDLE ENGINE RATING

- b. TAKEOFF, HOVER, OR LAND AT T/W = _____ FOR _____ HRS.
or: TAKEOFF, HOVER, OR LAND AT PEHF = _____, FOR _____ HRS.
- c. CLIMB TO _____ FT. WITH MAX R/C AT _____ ENGINE RATING
CLIMB TO _____ FT. WITH CONSTANT EAS AT _____ ENGINE RATING
CLIMB TO _____ FT. WITH CONST. MACH NO. AT _____ ENGINE RATING
CLIMB TO _____ FT. WITH CONSTANT TAS AT _____ ENGINE RATING
CLIMB TO OPT. ALT. FOR NEXT CRUISE WITH MAX. R/C AT _____ ENGINE RATING, MAXIMUM ALT. _____ FT.
CLIMB TO OPT. ALT. FOR NEXT CRUISE WITH CONSTANT EAS AT _____ ENGINE RATING, MAXIMUM ALT. _____ FT.
CLIMB TO OPT. ALT. FOR NEXT CRUISE WITH CONST. MACH NO. AT _____ ENGINE RATING, MAXIMUM ALT. _____ FT.
CLIMB TO OPT. ALT. FOR NEXT CRUISE WITH CONSTANT TAS AT _____ ENGINE RATING, MAXIMUM ALT. _____ FT.
- d. CRUISE AT _____ ENGINE RATING
CRUISE AT _____ KNOTS TAS LIMITED BY _____ ENGINE RATING
CRUISE AT BEST RANGE SPEED WITH HEADWIND OF _____ KNOTS
CRUISE AT SPEED FOR 99 PERCENT BEST RANGE WITH HEADWIND OF _____ KNOTS
CRUISE AT BEST RANGE SPEED WITH HEADWIND OF _____ KNOTS, CONSTANT W/DELTA = _____
- e. DESCEND TO H = _____ FT AT CONSTANT EAS
DESCEND TO H = _____ FT AT CONSTANT TAS
DESCEND TO H = _____ FT AT CONSTANT TAS (SPIRAL DESCENT PATH)
DESCEND TO H = _____ FT AT CONSTANT EAS (SPIRAL DESCENT PATH)
DESCEND TO H _____ FT AT CONSTANT MACH NO.
DESCEND TO H = _____ FT AT CONSTANT MACH NO. (SPIRAL DESCENT PATH)

DESCEND TO H = _____ FT., R = _____ NM AT CONSTANT EAS

DESCEND TO H = _____ FT., R = _____ NM AT CONSTANT MACH NO.

DESCEND TO H = _____ FT., R = _____ NM AT CONSTANT TAS

f. LOITER FOR _____ HRS.

g. CHANGE FUEL, ADD _____ LB

CHANGE FUEL, REMOVE _____ LB

h. CHANGE PAYLOAD, ADD _____ LB

CHANGE PAYLOAD, REMOVE _____ LB

i. TRANSFER ALTITUDE TO _____ FT

After the complete mission history has been printed, the following fuel summary will be printed:

MISSION FUEL REQUIRED =

RESERVE FUEL REQUIRED =

TOTAL FUEL REQUIRED =

NOTE: If segments 1 through 6 are used for reserve fuel calculations, headings a. through f. will be followed by the statement FOR RESERVE FUEL.

6.2 LIST OF DIAGNOSTIC ERROR PRINTOUTS

6.2.1 Errors Affecting Main Control Loop

- 6.2.1.1 *** ERROR THE USER REQUESTED PRIMARY ENGINE CYCLE NO. XXX BUT THE INPUT DECK WAS SET UP TO USE NO. YYY

The operator used an engine cycle whose identification number differed from that requested by the user (LOC 0217)

REMEDY: Use correct engine cycle

- 6.2.1.2 *** ERROR THE USER REQUESTED ROTOR MAP NO. XXX BUT THE INPUT DECK WAS SET UP TO USE NO. YYY

The operator used a rotor map whose identification number differed from that requested by the user (LOC. 0170)

REMEDY: Use correct rotor map

- 6.2.1.3 *** ERROR THE USER REQUESTED ROTOR CYCLE NO. XXX BUT THE INPUT DECK WAS SET UP TO USE NO. YYY

The operator used a rotor cycle whose identification number differed from that requested by the user (LOC. 0171)

REMEDY: Use correct rotor cycle

- 6.2.1.4 *** ERROR THE USER REQUESTED AUXILIARY ENGINE NO. XXX BUT THE INPUT DECK WAS SET UP TO USE NO. YYY

The operator used a auxiliary engine whose identification number differed from that requested by the user (LOC. 0242)

REMEDY: Use correct auxiliary engine

- 6.2.1.5 *** ERROR THE USER REQUESTED PROP TABLE NO. XXX BUT THE INPUT DECK WAS SET UP TO USE NO. YYY

The operator used a propeller table whose identification number differed from that requested by the user (LOC. 0260)

REMEDY: Use correct propeller table

- 6.2.1.6 ERROR *** THE FIRST SEGMENT INDICATOR OF A MISSION CANNOT BE 0., 100., or 5. (RMAXND = 0) SEE USERS MANUAL

a. Segment indicators 0., 100., represent the end of a mission calculation and the end of a particular case respectively. Either of these indicators at the beginning of a set would be meaningless.

b. Descent (RMAXND = 0) must be preceded by a cruise segment.

REMEDY: Rewrite segment indicator list (Starts at (LOC 0035))

6.2.1.7 ERROR *** DESCENT (RMAXND = 0) MUST BE PRECEDED BY A CRUISE

See 6.2.1.6 (b)

REMEDY: Redefine the mission with a different sequence of segment indicators

6.2.1.8 *** ERROR FUEL AVAILABLE AND FUEL REQUIRED DO NOT CONVERGE AT A POSITIVE GROSS WEIGHT

This indicates that the performance requirements are too stringent or that the weight is excessive. This may be due to pessimistic weight input constants or drag characteristics, or it may be that the mission required cannot be flown by any helicopter sized by HESCOMP. It may require some novel design considerations.

6.2.1.9 *** ERROR WFR WEIGHT OF FUEL REQUIRED IS LESS THAN OR EQUAL TO ZERO

This message can occur only if negative values of reserve fuel factors are input (LOC 0032, 33, 34)

REMEDY: Correct reserve fuel factors

6.2.1.10 ***** THIS AIRCRAFT HAS NOT CONVERGED AFTER 25 ATTEMPTS. THE WEIGHT OF FUEL AVAILABLE (WFA) = XXX THE WEIGHT OF FUEL REQUIRED (WFR) = YYY. (WFA) MUST BE WITHIN ZZZ OF (WFR) FOR THE AIRCRAFT TO CONVERGE. THIS TOLERANCE IS SET IN THE MAIN PROGRAM UNDER THE NAME TOL.

If this message occurs it is probable that the mission required cannot be flown by an aircraft of the type specified by the input data. A possible cause may be unrealistic input values of the reserve fuel factors or weights constants.

- 6.2.1.11 *** ERROR - NO TITLE CARD AFTER SEVEN CARD, COLUMNS 1 THROUGH 6 ON TITLE CARD MUST BE BLANK OR THERE WAS A 6 IN COLUMN 5 OF AN INPUT CARD

This message is printed if the input card deck is improperly set up. No output is generated.

REMEDY: Examine input deck for errors indicated in message and correct them.

- 6.2.1.12 J DOES NOT CONVERGE IN 25 ITERATIONS - SUBROUTINE POWER

This message is printed if a match cannot be found between propeller thrust and available power. The message is printed in forward flight calculations.

REMEDY: Increase engine power, reduce drag or modify propeller related inputs.

- 6.2.1.13 WARNING: THIS CASE HAS BEEN TERMINATED BECAUSE THE CALCULATED FUSELAGE CONSTANT DIAMETER SECTION (CABIN LENGTH) IS LESS THAN ZERO. CHECK ALL FUSELAGE AND ROTOR SIZING INPUTS.

This message will be printed if in the process of sizing a tandem rotor helicopter (FDMIND = 1) fuselage, l_c is calculated as a negative number.

REMEDY: Review input data that specifies tandem rotor helicopter fuselage size requirements.

- 6.2.1.14 WARNING: THIS CASE HAS BEEN TERMINATED BECAUSE THE CALCULATED TANDEM ROTOR OVERLAP RATIO EXCEEDS _____ CHECK ALL FUSELAGE AND ROTOR SIZING INPUTS

This message will be printed if in the process of sizing a tandem rotor helicopter (FDMIND = 3) fuselage, the overlap/diameter ratio exceeds the value printed in the error message.

REMEDY: Review input data that specifies tandem rotor helicopter fuselage size requirements.

- 6.2.1.15 AFTER 20 ITERATIONS, DTR DID NOT CONVERGE (SUBROUTINE SIZTR)

This message will be printed if the tail rotor diameter sizing iteration option (TRDIND = 3) does not converge.

REMEDY: Check all tail rotor sizing input data
((T/A)NET, b_{TR} , θ_{TTR} , X_{CTR} , etc.)

6.2.1.16 AFTER 20 ITERATIONS, X_M/l_B DOES NOT CONVERGE

This message will be printed if X_M/l_B calculated
by the main rotor position sizing option
(MRPIND = 1 or 2) does not converge.

REMEDY: Check all input values required for this
option (LOC 2678 - LOC 2696)

6.2.1.17 GAMMA FAILED TO CONVERGE IN 20 ITERATIONS -
SUBROUTINE THRUST

This message will be printed if the propeller
thrust calculation routine cannot converge on a
thrust and efficiency that will match the re-
quired thrust. Such an error is unlikely and
would occur only in cases involving extreme
thrust requirements.

REMEDY: Review input data that specifies propeller
requirements.

6.2.2 Errors Related to Tabulated Inputs

6.2.2.1 Two Dimensional Tables

*** ERROR***THE FOLLOWING VALUES MAY NOT BE ACCURATE.
THE INDEPENDENT VARIABLE WAS OUT OF RANGE OF THE
TABLE. THESE VALUES WERE CALCULATED USING THE YYYY
VALUE GIVEN IN THE TABLE. THIS ERROR IS IN THE XXXXX
TABLE.

This message occurs whenever the computer is
required to look up a value in a table of input
quantities at a calculated value of the indepen-
dent variable which lies outside the range of
the input values of the independent variable.

If the calculated independent variable is below
the lowest value of the input table, the computer
uses the first value in the table and YYYY in the
message reads FIRST. If it lies above the high-
est value, the last value in the table is used
and YYYY becomes LAST.

XXXXX in the last line of this message identifies
the table in which the error occurs. The tables
in which this could occur are shown below. The
third column indicates the part of the message
which is shown above as XXXXX.

<u>INDEPENDENT VARIABLE</u>	<u>DEPENDENT VARIABLE</u>	<u>XXXXX</u>
M	η_{p4}	M, ETAP4
C_L	$C_{D_{Wi}}$	CL, CDWI
h	θ	H-THETA
C_T	K_{HOVA}	CT, KHOVA TABLE (SUBR. ROTPOW) +
EPSILON	K_{INTO}	EPSILON-KINTO TABLE (SUBR. ROTPOW) ++
μ	K_{NUD}	MU-KNUD TABLE (SUBR. ROTPOW) ++
C_T	$K_{HOVA_{ATR}}$	CT, KHOVATR TABLE (SUBR. ROTPOW) ++
(O/L)/D	K_{2W}	SIZTR SUBR.
N_{II}/N_{IIOPT}	K_{PN}	NSUB2 CORRECTION FACTOR
$(N_I/N_I^*) (D/v_1)$	K_{PR}	REYNOLDS NUMBER
Q/Q^*	T/θ	TORQUE LIMIT LOOK UP
$(SHP/\delta \sqrt{\theta SHP^*})_{REQ}$	T/θ	POWER REQUIRED LOOK UP
C_L	γ	PROPELLER EQUIVALENT LIFT DRAG POLAR +
C_{Ti}	C_P	CTI- CP

+ Strictly speaking, this table is not an input. The table is calculated in the main control loop using BLOCK DATA and the input value of INTEGRATED LIFT COEFFICIENT (LOC. 0259). The error message indicates that the value of C_L used was above the maximum value in the table. This will occur only if an unusual combination of high power coefficient and low propeller activity factor exists. In such a case the user should change the propeller input parameters to obtain a propeller that more closely matches the performance requirements.

++This table is not input. It is stored as BLOCK DATA

6.2.2.2 Three Dimensional Tables

ERROR THE FOLLOWING VALUES MAY NOT BE ACCURATE. THE AAA INDEPENDENT VARIABLE IS OUT OF RANGE OF THE TABLE. THE PROGRAM USED THE BBB VALUE IN TABLE TO CALCULATE. THIS ERROR IS IN THE XXXX PART OF THE YYYYYY TABLE.

This message is printed whenever one of the calculated independent variables lies outside the range of the independent variables defining the table input by the user.

The XXXX and YYYYYY parts of the message respectively name the independent variable and the table in which the error occurred. AAA stands for FIRST or SECOND, BBB stands for First or LAST.

The tables in which this error could arise are shown below. The items in parentheses show the variables as they appear in the error message.

<u>DEPENDENT VARIABLE</u>	<u>INDEPENDENT VARIABLES (XXXX)</u>	<u>NAME OF TABLE (YYYYY)</u>
ΔC_{DM}	$C_L(CL), M$	COMPRESSIBILITY DRAG
$F_N/\delta F_N^*$	$T/\theta (T), M$	REFERRED THRUST
$SHP/\delta \sqrt{\theta} SHP^*$	$T/\theta (T), M$	REFERRED POWER
$\dot{W}/\delta \sqrt{\theta} SHP^*$	$T/\theta (T), M$	REFERRED FUEL FLOW
or		
$\dot{W}/\delta \sqrt{\theta} F_N^*$		
$N_I/\sqrt{\theta} N_I^*$	$T/\theta (T), M$	REFERRED NSUB1
$N_{II}/\sqrt{\theta} N_{II}^*$	$T/\theta (T), M$	REFERRED NSUB2
C_T	$C_p(CP), M$	PROPELLER POWER COEFFICIENT
C_T'/σ	$\mu, C_X/\sigma$	$CT'/SIGMA$ TABLE (SUBR. ROTLIM)

<u>DEPENDENT VARIABLE</u>	<u>INDEPENDENT VARIABLES (XXXX)</u>	<u>NAME OF TABLE (YYYYY)</u>
$\Delta K_{HOV} \theta_T$	C_T, θ_{TMR} or C_T, θ_{TTR} of C_T, θ_{TREF}	DELTA KHOVER - THETA T TABLE (SUBR ROTPOW) †
σ_{IF}	S, Z_W	PRANDTL DRAG INTER- FERENCE TABLE (AERO SUBR) †
C_P/σ	$C_T'/\sigma, C_X/\sigma$	CTP/SIGMA ROTOR MAP
C_{P_H}/σ	C_T, M_{TIP}	HOVER PORTION ROTOR MAP

REMEDY: Rewrite the input table so that the independent variable that was previously out of range will fall into the range of the new table.

†This table is not input. It is stored as BLOCK DATA.

6.2.3 Errors Occuring in Performance Calculations

6.2.3.1 WARNING: ROTOR LIMIT HAS BEEN EXCEEDED. FORWARD FLIGHT SPEED HAS BEEN REDUCED ACCORDINGLY. CHECK ALL VALUES OF TAS, MU, CT'/SIGMA AND CXR IN THIS PERFORMANCE LEG.

This message will be printed in segments climb, cruise, descent and loiter if a rotor limit has been exceeded.

REMEDY: Check rotor limits tabular input values (LOC 0347 - 0395) and segment input values.

6.2.3.2 WARNING: ROTOR LIMIT HAS BEEN EXCEEDED. EITHER REDUCE MAIN ROTOR THRUST REQUIREMENTS AT THESE OPERATING CONDITIONS OR INCREASE MAIN ROTOR TIP SPEED, CHECK ALL VALUES OF CT/SIGMA IN THIS PERFORMANCE LEG.

This message will be printed in the takeoff hover, and landing segment if a rotor limit has been exceeded.

REMEDY: Check rotor limits tabular input values (LOC -347 - 0395) and segment input values.

6.2.3.3 CAUTION: TAIL ROTOR ANTI-TORQUE THRUST REQUIRED AT THIS OPERATING CONDITION IS NEGATIVE. CHECK ALL VALUES OF CLFIN.

This message is printed in forward flight (climb, cruise, descent, and loiter segments) when single rotor helicopter vertical tail fin lift exceeds the total anti-torque thrust required at a given operating condition (resulting in a negative anti-torque thrust required for this tail rotor). This condition can be the result of:

- a) too much vertical tail fin area
- b) tail fin operating at too large a fin C_L

REMEDY: Check size requirements for vertical tail and input value for fin operating C_L (LOC 0216)

6.2.3.4 CAUTION: TAIL ROTOR CT EXCEEDS .030. TAIL ROTOR TIP SPEED RESET - CHECK ALL VALUES IN PERFORMANCE LEG.

This message is printed when the tail rotor C_T exceeds .03. This condition can be the result of

- a) too small a tail rotor diameter
- b) operating the tail rotor at too low a tip speed

REMEDY: Check the sizing requirements for the tail rotor diameter, or increase the tail rotor tip speed.

6.2.3.5 INSUFFICIENT POWER AVAILABLE TO HOVER. (T/W) AVAILABLE LESS THAN (T/W) REQUIRED AT DESIGN DOWNLOAD.

This message will be printed out during the take-off, hover and landing segment if the value of T/W calculated from the input PEHF (TOLIND = 2 or 4) is less than that required to provide sufficient thrust to overcome hover download (based on the design (T/W)_D LOC 0228). NOTE: (T/W)_D is always input.

REMEDY: Increase the input value of PEHF

6.2.3.6 CAUTION ** PEHF IS GREATER THAN 1

This message indicates a condition for which greater than 100 percent of maximum power or thrust was required during takeoff, hover or landing.

REMEDY: Increase engine power available or decrease required thrust-weight for hover

6.2.3.7 WARNING: THIS CASE HAS BEEN TERMINATED BECAUSE OF INSUFFICIENT POWER AVAILABLE FOR CLIMB AT THIS FLIGHT CONDITION. CHECK ALL INPUTS

This message is printed if the engine thrust or power input by the user is insufficient to allow the aircraft to climb.

REMEDY: a) Increase the engine power or thrust, whichever is appropriate, or
b) Inspect the inputs which determine drag and adjust them if they appear to give a grossly over-rated value of drag.

6.2.3.8 WARNING: THIS CASE HAS BEEN TERMINATED BECAUSE THE CLIMB ANGLE IS TOO LARGE DUE TO EXCESSIVE POWER AVAILABLE AT THIS FLIGHT CONDITION. CHECK ALL INPUTS

This message is printed if the engine thrust or power input by the user is excessive (resulting in climb angles greater than 45°) for the flight condition desired.

REMEDY: a) Decrease the engine power or thrust or
b) Increase the value of the drag input to this segment

6.2.3.9 INSUFFICIENT POWER AVAILABLE FOR CRUISE AT INPUT TAUXTT

This message will be printed during cruise, if in the case of a compound or auxiliary propulsion helicopter, the input value of T_{AUX}/T_{TOT} does not permit a power or thrust available-required match at a given cruise speed.

REMEDY: Check T_{AUX}/T_{TOT} inputs in the cruise segment.

6.2.3.10 ERROR ***** INSUFFICIENT POWER AVAILABLE FOR CRUISE
AT DESIRED SPEED

This message will be printed during cruise if CRSIND = 2 (cruise at specified true airspeed) and insufficient power is available to maintain steady level flight at the desired speed. The remaining cruise calculations will be at constant power setting.

REMEDY: Increase engine power, decrease drag level, or decrease required cruise speed.

6.2.3.11 CAUTION SPEED LIMITED BY POWER/THRUST AVAILABLE AT
SPECIFIED POWER SETTING

This message will be printed when power or thrust available is insufficient to allow the aircraft to cruise at speed for 99% best range as specified by selecting CRSIND = 4. or 6. (LOC 0721 through 0730)

6.2.3.12 INSUFFICIENT POWER FOR STEADY LEVEL FLIGHT

This message appears during the loiter segment calculations.

REMEDY: Check power available and drag level.

6.2.3.13 CLQS IS TOO LARGE FOR DESCENT AT REQUIRED SPEED

This message is printed out when, in the case of a winged helicopter, the wing contribution to the total aircraft lift is too high to permit this aircraft to descend.

REMEDY: Reduce wing operating C_L

6.2.3.14 INSUFFICIENT POWER TO DESCEND AT THE REQUIRED SPEED

This message is printed out when the aircraft has insufficient power to descend at the required speed.

REMEDY: Reduce value of drag input in this segment.

6.2.3.15 TERMINAL RANGE EXCEEDED, SPIRAL DESCENT REQUIRED

This message is printed when the predicted flight path ends beyond the specified terminal range (RMAXND = 1).

REMEDY: Reevaluate terminal range requirement or increase value of drag input in this segment.

6.2.3.16 TERMINAL RANGE NOT ATTAINED, USE MORE CRUISE OR A SHALLOWER DESCENT

This message is printed out when the predicted flight path ends before the specified terminal range ($R_{MAXND} = 1$).

REMEDY: Reevaluate the terminal range requirement or reduce the drag input to this segment.

6.2.3.17 **ERROR** THE RANGE NECESSARY TO DESCEND IS GREATER THAN THE RANGE OF THE TABLE CALCULATED IN CRUISE. THIS MAY BE DUE TO A DELTA R IN CRUISE WHICH IS TOO SMALL.

The computer saves the last ten points of the cruise from R_{max} backward so that an iteration can be carried out to find the correct point to start the descent when $R_{MAXND} = 0$. This error message is printed if the stored values do not cover a sufficient range back from R_{max} . Either ΔR is too small or the angle of descent is very small.

REMEDY: Check ΔR (LOC. 0771 through 0780).

7.0 PROGRAM USAGE

7.1 COMMENTS ON PROGRAM USAGE

Following are a list of rules and suggestions for using the program:

7.1.1 Rules

1. Do not use descent option $RMAXND = 0$ unless preceded by a cruise.
2. Do not input a turbofan or turbojet engine cycle for the primary engines.
3. $(T/W)_D$ (LOC 0228) must always be input. This is the basic configuration design thrust-to-weight ratio. It is used to establish the basic configurations download for calculation of both hover and low forward speed performance.
4. If $FIXIND = 0$ and $FIXINDI = 1.0$, locations 0234 - 0241 must be input to allow sizing of the auxiliary independent engines.
5. If $FIXIND = 1$ and $ESCIND = 1$ only fixed size auxiliary independent engines ($FIXINDI = 0.0$) may be input.
6. If $OPTIND = 2$, the helicopter parasite drag should be input as two terms. The wing (if there is one) profile drag coefficient is input to the table of C_{Dwi} versus C_L , and all other component contributions are input by means of the term ΔF_e (LOC 0316). The terms C_{DAP} , C_{DFP} , C_{DCSMR} , C_{DSHMR} , C_{DCSTR} , C_{DSHTR} , C_{DN} , C_{DNI} , C_{DNS} , C_{DVT} , and C_{DHT} . K_{HPIM} , K_{HPIT} , K_N , K_{NI} , K_{NS} , K_F , K_{VT} , and K_{HT} are not used in $OPTIND = 2$. If the option indicator is 1, all terms and factors may be used.
7. If cruise is followed by descent with $RMAXND = 0$, the cruise step size (LOC. 0771 - 0780) should not be less than 10 to 15 nautical miles. This is necessitated by the fact that a table of cruise conditions is compiled during cruise to use in the determination of the starting point for descent. This table consists of 10 points. The cruise step size therefore must be sufficiently large to ensure that the total of nine steps in range is greater than the range required for the following descent. A cruise step size which is too small will lead to termination of the case with the printout:

*** ERROR *** THE RANGE NECESSARY TO DESCEND IS GREATER THAN THE RANGE OF THE TABLE CALCULATED IN CRUISE. THIS MAY BE DUE TO A DELTA R IN CRUISE WHICH IS TOO SMALL.

8. At present do not use SGTIND = 7 with OPTIND = 1 unless a sufficiently large ΔW_f is input to completely refuel the aircraft. This rule will be eliminated by future modifications to the program. For the present, missions employing change of fuel can be analyzed by running separate cases, a new case each time the fuel is changed. The aircraft can be separately sized for each case and compared manually.
9. The value for payload which is input (LOC. 2604) should be the payload at initial takeoff.

7.1.2 Suggestions

1. Input locations 0005 → 0022 are arranged in a sequential order (1st and 2nd order size trend and propulsion indicators) which allows configuration types to be input in a logical "building block" manner. Use of this arrangement facilitates the input of data. For example if the user wishes to input a single rotor auxiliary independent engine compound helicopter, the following input sequence follows:
 - a) Input CNFIND = 1 (single rotor helicopter)
 - b) Input AUXIND = 4 (compound helicopter)
 - c) Since a compound helicopter has wings, input desired wing sizing options (S_w IND and b_w IND).
 - d) Input AIPIND = 2 (auxiliary independent engines)
 - e) Input desired type of auxiliary independent engine (ENGIND)
 - f) Input those options pertaining only to single rotor helicopters (TRDIND, TRSIND, VTFIND, HTIND, and MRPIND)
2. If nonstandard atmosphere is required only for constant altitude segments, such as loiter, cruise, and takeoff, the table of temperature ratio versus altitude need not be filled in. The nonstandard atmosphere may be obtained by use of ATMIND = 1.
3. If it is desired to run OPTIND = 2 for a helicopter which has previously been sized in a separate case, the drag

will be represented correctly if the output values of a_5 and a_6 from the sizing case are input for the OPTIND = 2 case to ΔF_e (LOC 0316) and K_W (LOC 0327) respectively, and if the C_{Dwi} table is filled in identically to the sizing case.

4. To represent engines which are buried in the fuselage, input K_N (LOC 0323) or K_{NI} (LOC 0324), or both as zero and Z_1, Z_2, Z_3 (LOCS 0142, 0143, and 0144) or Z_4, Z_5, Z_6 (LOCS 0146, 0147, 0148) or both as zero. The component drag will then be zero and the calculation of engine/nacelle dimensions will be bypassed.
5. The order in which segments 7 and 8 are used is important due to the fact that the program will not permit the aircraft weight, during a change of weight segment, to exceed gross weight. As an example, to simulate adding 200 pounds of payload, followed by refueling back to gross weight limits, the eighth performance segment (change of payload weight) should be entered first with an input of ΔW_{PL} (LOC 1161 through 1170) of 200. Then, the change of fuel weight segment can be entered with a large number input for the ΔW_f quantity.
6. The weights factors K_1 through K_{20} (LOC 2654-2673) have a nominal value of 1.0 assigned to them by the program. These factors need not be input unless a nonunity value is desired. Similarly, the incremental group weights, ΔW_{FC} (LOC 2605), ΔW_p (LOC 2606), and ΔW_{ST} (LOC 2607), are nominally zero and need not be input unless a nonzero value is desired. The reserve fuel factors K_1 (LOC 0032) and δW_f (LOC 0033) are nominally unity and zero respectively. The fuel flow multiplier K_{FF} (LOC 0034) is nominally 1.0.
7. A cruise may be run with a headwind for cruise options 3 through 6 by input of the headwind in knots in locations 0731 through 0740. For cruise option 2 (specified constant true airspeed), the user can simulate cruise with a headwind by inputting an "equivalent" value for R_{max} (LOC 0791 - 0800), obtained by adjusting the true ground range desired by the ratio airspeed $\div$ ground speed. The program output values for range must then be readjusted by the inverse of this ratio to obtain the correct ground range.

7.2 DISCUSSION OF ROTOR PERFORMANCE CALCULATIONS

As noted in section 4.5, three options for computing rotor performance are available to the program user. The first of these (ROTIND = 1) utilizes the "short form aero" rotor performance method which can be described as follows:

The short form aero rotor performance methodology is a combination of momentum theory and empirically derived factors. The four elements of the rotor power required are:

- a) induced power (power required to generate lift)
- b) profile power (power required to turn the rotor)
- c) parasite power (power required to supply propulsive thrust in forward flight).
- d) nonuniform downwash power (power correction due to nonuniform inflow and downwash effects in forward flight).

Figure 7-1 is a summary of the major equations used in this methodology. A brief description of their applications follows:

In hover, the rotor power required is composed of only two parts, induced and profile power. The induced power as represented by the equations in Figure 7-1 is a function of the variables K_{HOV} , K_{OL} , and C_T . K_{HOV} is the adjustment for non-uniform inflow and wake contraction effects and is a function of C_T , blade number, and blade twist. K_{OL} is the correction for overlapping rotors (as in the case of a tandem rotor helicopter).

The profile power is simply a function of the integrated blade drag coefficient (including compressibility effects) at a specified operating C_T/σ and blade solidity.

In cruise, the rotor power is composed of all four of the components listed initially. The induced power, as represented by the equations in Figure 7-1, is a function of the quantities K_{IND} , K_{INT} , C_T' , and μ' . K_{IND} is the induced power adjustment factor which accounts for blade tip and other losses. K_{INT} is the induced power adjustment for interference between tandem rotors. Thus, for single rotor helicopters, K_{INT} is equal to 1. For tandem rotors, the value of K_{INT} is calculated based on tandem rotor overlap and an empirically derived wake separation angle, ϵ' . Profile power is simply a function of the integrated blade drag coefficient (corrected for retreating blade stall and advancing blade compressibility effects) at specified operating conditions (C_T'/σ , μ , C_x), blade solidity, and advance ratio (μ). The parasite power is a function of the propulsive thrust required and the efficiency of the rotor in converting power into that propulsive thrust (in addition to providing lift). The nonuniform downwash (NUD) power is a correction which has been empirically derived from a comparison of uniform and nonuniform downwash rotor analyses. The term K_{NUD} , which is a function of the advancing rotor, is stored as BLOCK DATA in this program.

MAIN ROTOR IN HOVER SUMMARY	MAIN ROTOR IN CRUISE FLIGHT SUMMARY
$RHP_{TOT} = \frac{\rho AV_T^3 C_{PTOT}}{550}$	$RHP_{TOT} = \frac{\rho AV_T^3 C_{PTOT}}{550}$
where: $C_{PTOT} = C_{PPRO} + C_{PIND}$	where: $C_{PTOT} = C_{PPRO} + C_{PIND} + C_{PPAR} + C_{PNUD}$
(PROFILE POWER) $C_{PPRO} = C_{D_O} \frac{\sigma}{8} (1 - X_C)$	(PROFILE POWER) $C_{PPRO} = \frac{C_{D_O} \sigma}{8} (1 + 4.65 \mu^2) (1 - X_C)$
(INDUCED POWER) $C_{PIND} = .707 K_{HOV} K_{OL} C_T^{3/2}$	(INDUCED POWER) $C_{PIND} = \frac{K_{IND} K_{INT}}{2 \mu^2} C_T^{1.2}$
$C_T = \frac{\text{HOVER THRUST REQ'D}}{\rho AV_T^2}$	(PARASITE POWER) $C_{PPAR} = \mu C_X (K_{PER})$
$A = \frac{\pi D^2}{4} N_{ROT}$	(NUD POWER) $C_{PNUD} = \frac{2 K_{NUD} C_T' \sigma}{B^2 (1 + D.L. \sin^2 \epsilon)}$
$X_C = 2 r_{cutout} / D$	$C_T' = \frac{L_{ROT}}{\rho AV_T^2} (1 + D.L. \sin^2 \epsilon)$
	$C_X = \frac{X_{TOT}}{\rho AV_T^2} \quad A = \frac{\pi D^2}{4} N_{ROT}$
	$\epsilon = \tan^{-1} (2V / V \times 1.689)$
	$K_{PER} = \frac{\mu_{PRI}}{\mu_{PRR}} = 1 + 12.8 \mu^4$
	$v_i = V_T \sqrt{[(\mu^4 + C_T^2)^{1/2} - \mu^2] / 2}$
	$\mu' = \frac{\sqrt{(1.689V)^2 + v_i^2}}{V_T} \quad K_{NUD} = f(\mu)$

Figure 7-1. Short Form Aero Rotor Performance Equations Summary.

In order to obtain a reasonable estimation of power required at very low advance ratios ($\mu < 0.1$) where neither normal cruise nor hover rotor characteristics totally describe the operating environment of the rotor, an empirical fairing technique is used. The method is based on the contracted induced wake angle ϵ :

$$\epsilon = \tan^{-1} \left(2v_i / (1.689V) \right)$$

The relationship:

$$\sin^2 \epsilon + \cos^2 \epsilon = 1$$

is used to provide a smooth transition between hover and cruise characteristics for the affected coefficients while insuring that the resulting values will lie within the boundaries set by hover and cruise limiting conditions.

A detailed description of the equations used in this methodology is provided by inspection of Figure 4-14 (subroutine ROTPOW flow chart) and the input variable list included in paragraph 5.3.1 of Section 5.0. The empirical factors used in this methodology are input as noted earlier, in "rotor cycle" format. The input sheet used for this purpose is included in the specimen input sheets of Section 5.2. It should be noted that since the factors specified in a "rotor cycle" represent integrated blade characteristics, then a given "rotor cycle" implicitly represents a given spanwise chord and airfoil distribution. Thus, it would ultimately be possible to build up an extensive library of "rotor cycles" with varying combinations of planform and airfoil distributions.

In the case of ROTIND = 2 (as noted in Section 4.5) input rotor maps, corrected by the program for the specific rotor and helicopter configuration characteristics under study, are employed. When ROTIND = 3, input rotor maps with no corrections applied are utilized. A detailed description of the equations and variables used for ROTIND = 2 or 3 is available by inspection of Figure 4-14 and paragraph 5.3.1. Figures 7-2 and 7-3 show the corrections (K_{DLD} , TIGE/TOGE) for hover-in-ground effect applied to the equations:

$$T/W = 1 + D.L. (K_{DLD})$$

$$C_T = \frac{4W(T/W)}{\rho \pi D_{MR}^2 N_R V_{TIP}^2 (TIGE/TOGE)}$$

used in calculating hover power.

$$\text{DLD}_{\text{CORR.FACT}} = 1.028 - 1.25e^{-(2h_{B_F}/\text{DIA.})}$$

SINGLE
ROTOR

$$\text{DLD}_{\text{CORR.FACT}} = 1.36771 - 1.57913e^{-h_{B_F}/\text{DIA.}}$$

TANDEM
ROTOR

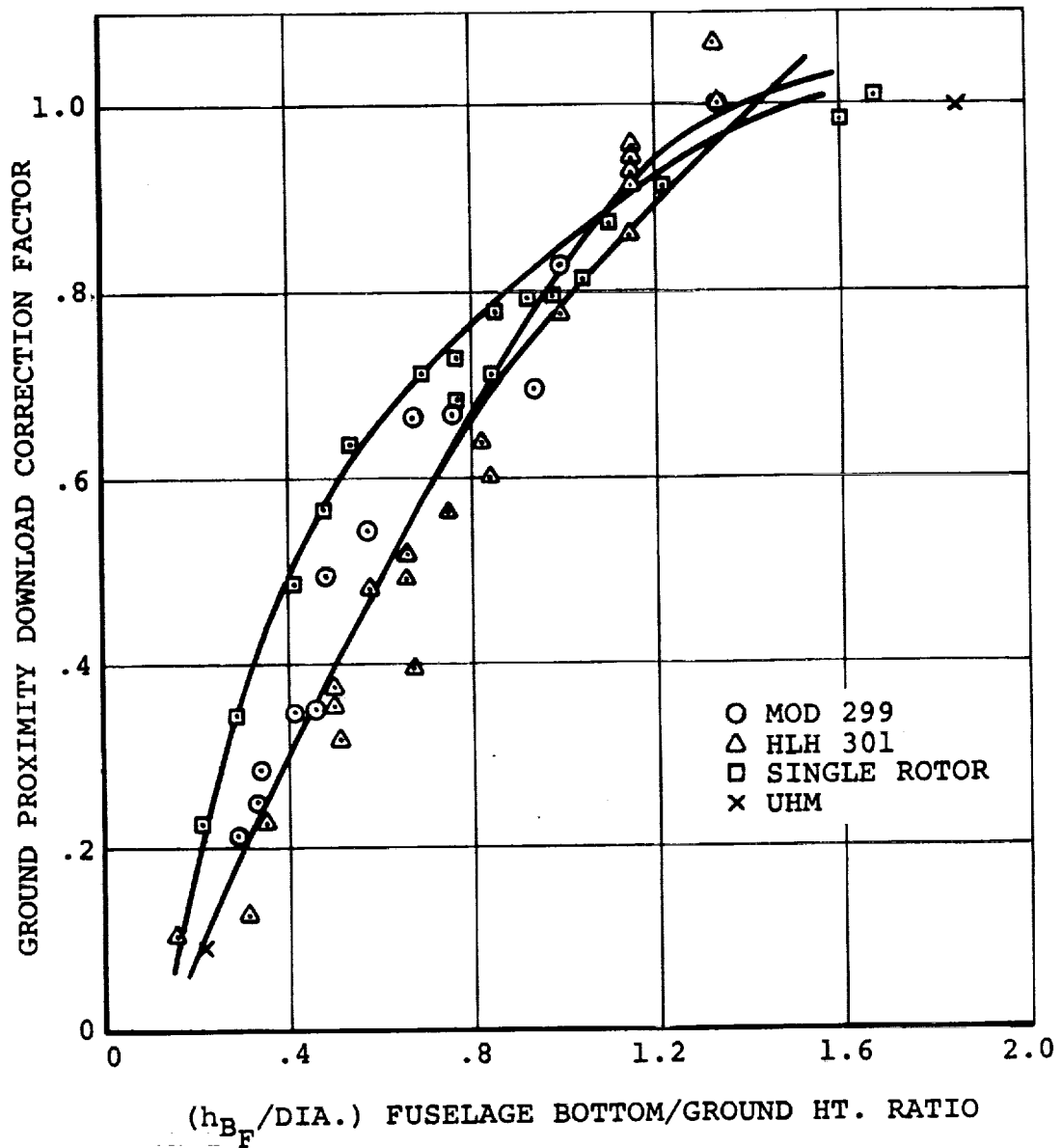


Figure 7-2. Download Sensitivity to Ground Proximity.

$$\text{THRUST}_{\text{IGE}}/\text{THRUST}_{\text{OGE}} = .06907(1/X) + .03364(X) + .90186$$

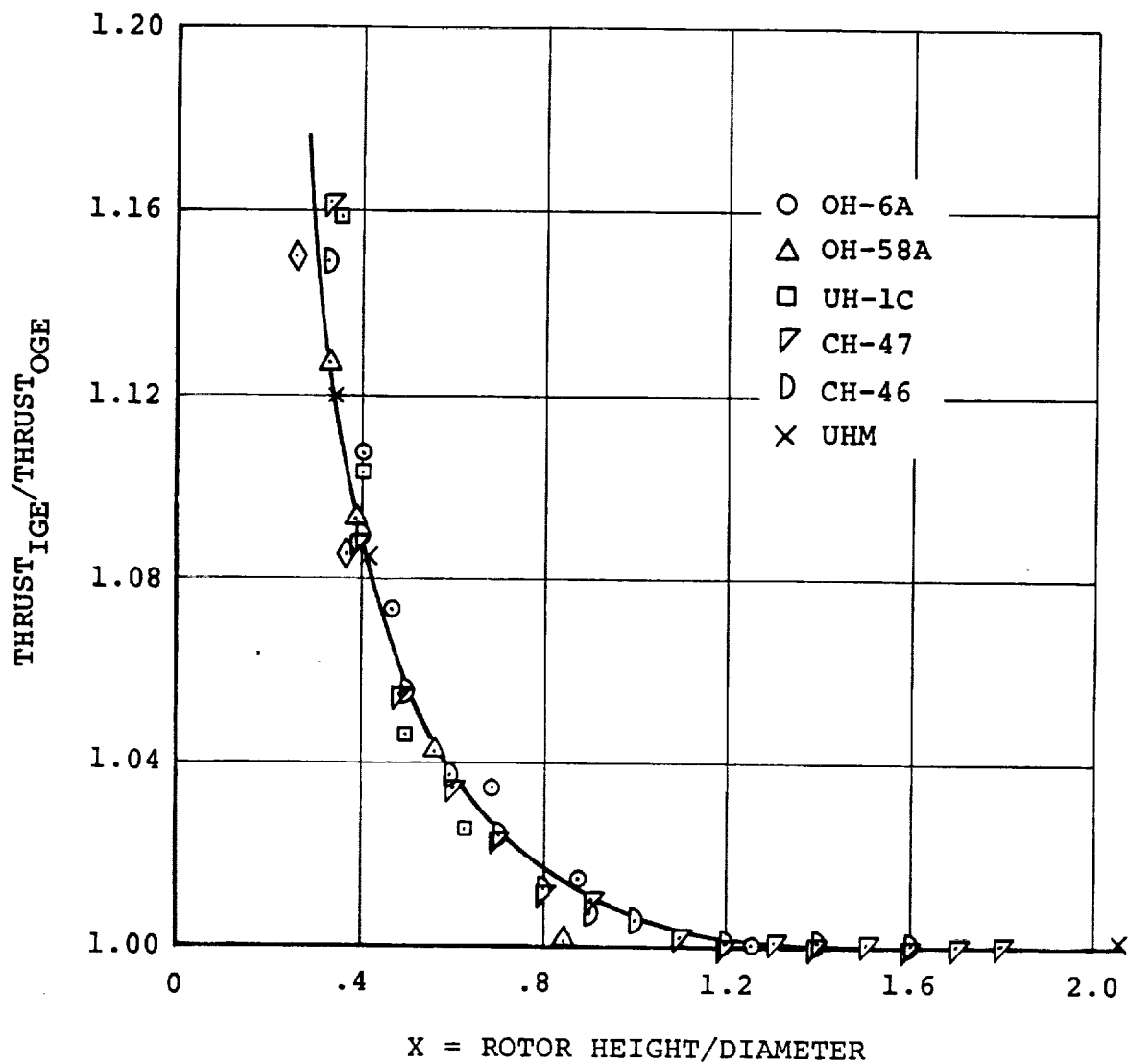
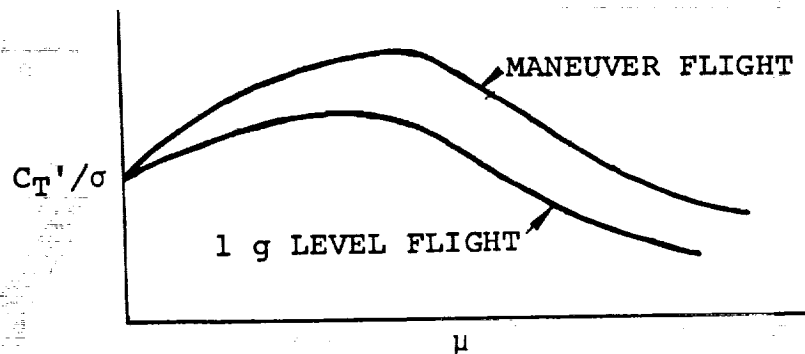


Figure 7-3. Thrust Augmentation in Ground Effect.

7.3 DISCUSSION OF ROTOR LIMITS

The function of the rotor limits table input described in Section 4.6 is to provide realistic (1g) level flight boundaries for helicopter rotor operation. This is important because, although the rotor performance calculation (whether using rotor "cycles" or maps) reflects operation near stall through rapidly increasing power required levels, it would still be possible, using a greatly oversized engine, to operate in this region, even though in actual fact the rotor could be overstressed or subject to structural failure. A typical rotor limits plot is illustrated by the sketch below:



For purposes of defining the tabular rotor limits input, the level flight conditions are of interest only, although single point values $(C_T/\sigma)_H$, $(C_T/\sigma)_{CR}$ from the maneuver flight curve are necessary for determining (sizing) main rotor solidity. Rotor limits, then can be based on:

- a) incipient rotor blade stall limits (1g level flight), or
- b) incipient rotor blade stall and/or rotor blade structural limits for maneuver flight.

Figure 7-4 shows a summary of miscellaneous rotor limits data (theoretical and flight test), for both the 1g level flight and maneuver conditions. The rotor limit value $(C_T/\sigma)_H$ encountered in hover is typically due to stall flutter. This is primarily an aeroelastic/control system stiffness problem. Level flight rotor limit values, as noted earlier, are a function of incipient stall and/or stall flutter. Rotor limits in maneuver flight are more complex to understand because of the interaction of various rotor configurations and rotor parameters on the result. For example, a rotor system with

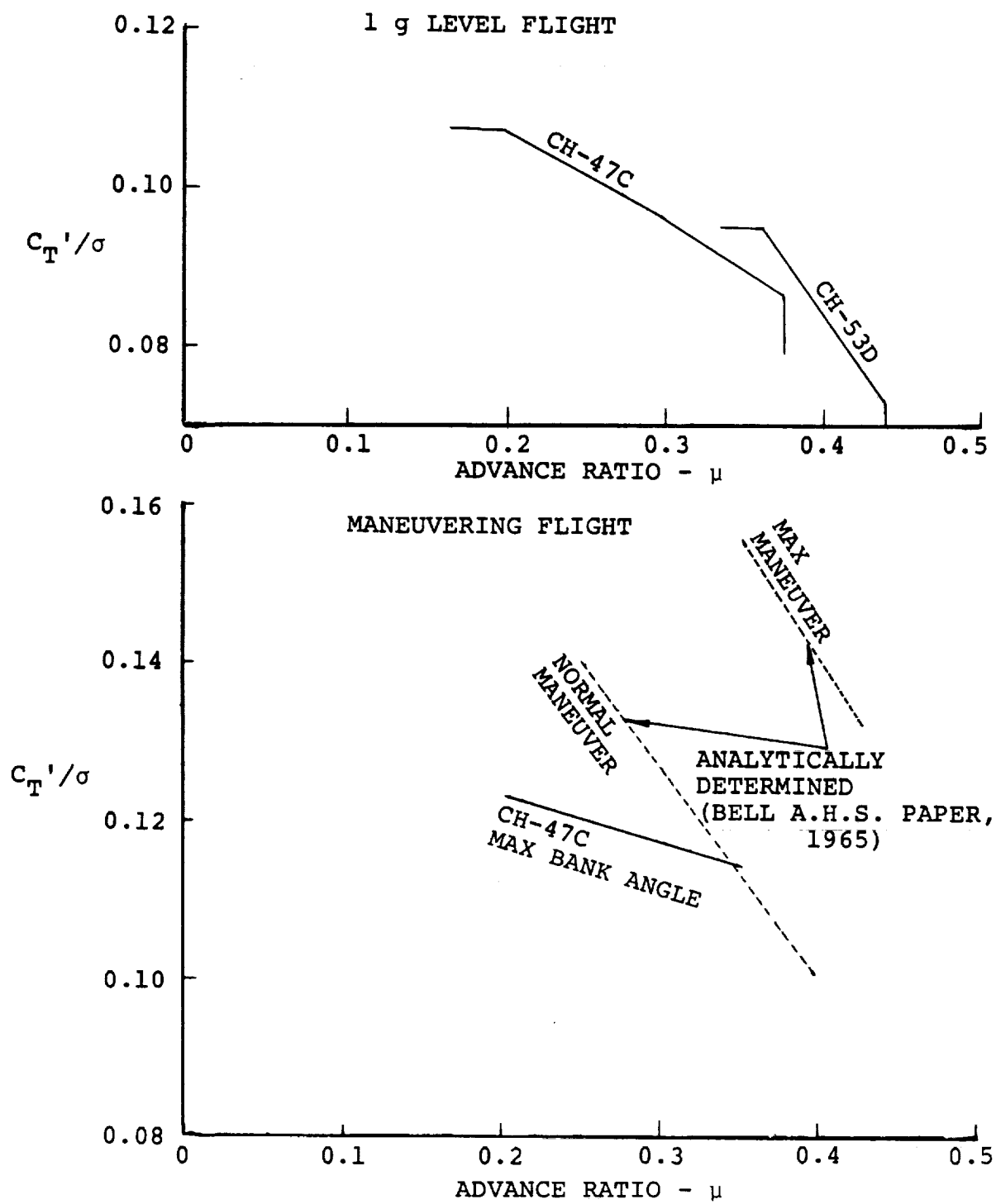
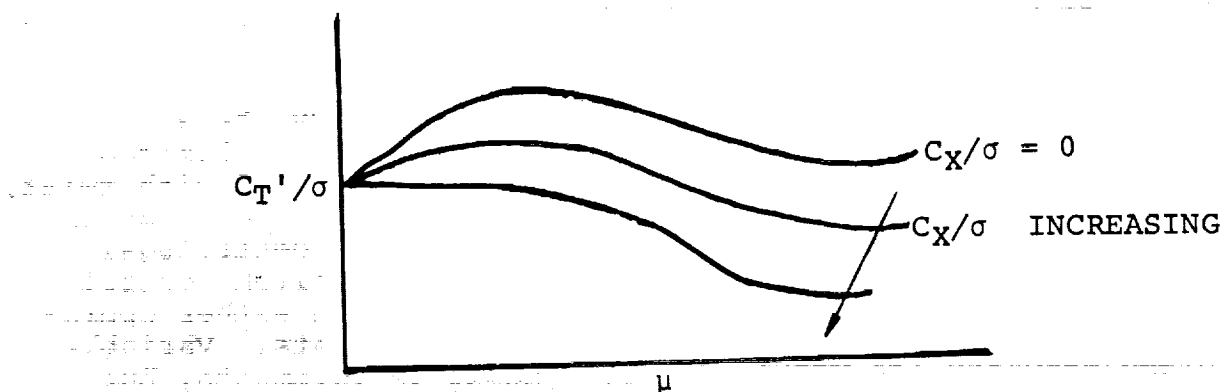


Figure 7-4. Summary of Typical Rotor Limits.

relatively high rotor blade inertia in the flapwise direction, should potentially (in a maneuver) exhibit a higher maneuver g capability (due to gyroscopic precessional effects) than a rotor with less inertia in the flapwise direction. Other factors influencing rotor limits include the torsional natural frequency of the blade as it interacts with stall flutter, chordwise bending stresses of the blade, the type of maneuver performed, etc. For a more detailed discussion of this matter see References 12 to 15.

Provision has been made in the rotor limits table for inclusion of rotor limits which are a function of C_X/σ (based on rotor propulsive thrust) as well as μ (see the sketch below).



In those instances where C_X/σ is not a variable, the user simply inputs C_T'/σ versus μ at dummy values of C_X/σ (0 and 1.0). If the user wishes to operate the program without using rotor limits, large "dummy" values of C_T'/σ (say 1.0) are input at $C_X/\sigma = 0$ and 1.0.

7.4 DISCUSSION OF PROPELLER EFFICIENCY

The final selection of a propeller blade design to best suit a given compound helicopter mission is a rather arduous task because the suboptimization of many considerations, such as propeller efficiency, propeller weight, power transmission system weight, powerplant performance, and others, is required for each mission segment followed by an overall mission optimization. A single propeller design does not satisfy the requirement.

The basic problem faced in evolving a single propeller design to satisfy all flight conditions is that of achieving the optimum blade loading for each of the flight conditions. This

is virtually impossible due to the degree and manner in which thrust required and power available vary with engine and vehicle speeds. From an aerodynamic viewpoint, this basic problem manifests itself in terms of problems associated with blade chord, twist and design C_L distributions, engine-propeller performance matching, and compressibility.

Propeller blade loading is a function of the spanwise distribution of blade twist, blade chord, and blade section design lift coefficient. These three parameters must be employed so as to yield the optimum propeller performance at a given flight condition. This will occur when each section of the blade is adjusted to operate at or near its maximum lift-drag ratio while maintaining an optimum spanwise load distribution. As the operating conditions vary, the degree to which near optimum conditions can be maintained changes for a fixed blade geometry. Therefore, some compromise must take place, and best efficiency cannot be achieved at each and every operating condition.

As one can appreciate, with fixed blade geometry the attainment of overall propeller optimization is somewhat limited with regard to what can be aerodynamically achieved with twist, solidity, and design lift coefficient. Furthermore, changing these variables results in variations in blade centrifugal twisting moment, hub centrifugal loads, blade pitch control loads, and numerous other items which result in either operational envelope limitations or weight constraints. Variable blade geometry can result in aerodynamic improvements, but these may well be offset by increased weight and cost. Variable geometry propeller blade development and application, furthermore, have been quite limited.

The ability to alter propeller speed in cruise will help the designer cope with blade loading problems and result in better mission efficiency. This can be done whether by using a multiple speed power transmission system between the engine and propeller or by exercising the variable output shaft speed capability of free turbine powerplants. The former method is generally not used due to weight penalties, while the latter method is extensively employed. Engine-propeller matching, though, is not as simple as it may sound. Engine power does fall off at nonoptimum turbine speed, and transmission torque requirements and weight increase with reduced turbine speed.

The combination of vehicle speed, propeller speed, diameter and altitude produce a constraint in the form of Mach number. Exceeding a helical tip Mach number of about 0.95 appears to significantly reduce propeller efficiency.

Current state of the art regarding propeller aerodynamics appears to permit very accurate appraisal of a given propeller design performance over most of the flight envelope. Performance prediction capability is generally inadequate in the following areas: 1) static thrust, 2) at moderate to high propeller shaft angles of attack (say 30 to 90 degrees), and 3) under the "mixed" flow conditions where the blade sections are in neither wholly subsonic nor wholly supersonic flows. For purposes of preliminary design, however, the short methods for predicting propeller performance available from propeller manufacturers (e.g., Curtiss-Wright and Hamilton Standard) generally produce acceptable results, and should certainly be given consideration.

Whenever possible, the aircraft designer should consult the propeller manufacturers' and his own propeller staffs early in the preliminary design phase. Lacking this, he should freely exercise the methodology published by propeller manufacturers. These methods require only several minutes to manually compute a propeller performance point and are well worth the effort. Too many preliminary aircraft designs have proceeded too far assuming propeller efficiencies in excess of the ideal induced (i.e., zero drag) value.

7.5 DISCUSSION OF PROGRAM TOLERANCES

The tolerances tabulated in Table 7-1 represent the accuracy required of iterated values calculated at certain points in the program. Whenever the values of the quantities named in Table 7-1 become less than the value quoted, the iterating calculation is terminated.

TABLE 7-1. PROGRAM TOLERANCES

SYMBOL	VALUE	VARIABLE BEING CALCULATED	SITUATION IN PROGRAM	FUNCTION OF TOLERANCE
TOL	0.01	W_G , Gross Weight	Main Control Loop	When the quantity $ 1 - (W_f)_A / (W_f)_R \leq \text{TOL}$, the fuel required and available are considered to be sufficiently close and the sizing calculation is terminated.
$\Delta\gamma$	0.1°	γ , Flight Path Angle	Climb & Descent Sub routines	Determines flight path angle to within 0.1°
$\frac{\Delta \text{BHP}}{\text{BHP}_A}$	0.01	$\frac{\text{BHP}_A - \text{BHP}_R}{\text{BHP}_A}$	Cruise	The cruise speed is set when BHP_R is within 0.01 BHP_A
ΔB	0.01	$\frac{B_1 - B_2}{B_2}$	CRSIND = 1	$\frac{B_1 - B_2}{B_2}$ is used to adjust ΔV to expedite computation. If $\frac{B_1 - B_2}{B_2}$ becomes less than ΔB , BHP_R always exceeds BHP_A
$\frac{\Delta (X_M / \ell_B)}{(X_M / \ell_B)_C}$	0.01	$\frac{(X_M / \ell_B) - (X_M / \ell_B)_C}{(X_M / \ell_B)_C}$	Main control loop	The main rotor position is determined when W_M / ℓ_B is within 0.01 $(X_M / \ell_B)_C$
$\frac{\Delta D_{TR}}{D_{TRI}}$	0.01	$\frac{D_{TR} - D_{TRI}}{D_{TRI}}$	Size trends subroutine	The tail rotor diameter is determined when D_{TR} is within 0.01 D_{TRI} .
$\frac{\Delta \sigma_{TR1}}{\sigma_{TR}}$	0.01	$\frac{\sigma_{TR1} - \sigma_{TR}}{\sigma_{TR}}$	Main control loop	The tail rotor solidity is determined when σ_{TR} is within 0.01 σ_{TR} .
R_{TOL}	5 nm R, Range		Descent Sub routine	If the range at the end of descent is within R_{TOL} nm of R_{max} the calculation terminates.

7.6 SAMPLE CASES

To illustrate the use of the program, two sample cases have been run and the output included here.

The first case is for a single-rotor compound helicopter with auxiliary independent cruise propulsion (T/Shaft - Propeller). This case illustrates main rotor (diameter and solidity) sizing, wing sizing for maneuver conditions, auxiliary independent engine sizing, tail rotor solidity sizing to meet hovering turn requirements, vertical tail area sizing based on tail rotor loss (in cruise) criteria, and the use of a drag trend. The primary engines and drive system are sized to meet specified takeoff and cruise requirements.

The second case is for a tandem rotor winged helicopter. It illustrates the use of the component drag buildup option, fuselage sizing based on specified rotor overlap and cabin dimensions, aft rotor pylon sizing based on an input gap/stagger ratio, wing sizing for maneuver conditions, and main rotor (diameter and solidity) sizing. The primary engines and drive systems are sized to meet specified takeoff and cruise requirements.

7.6.1 Single Rotor Compound Helicopter (Auxiliary Independent Engines)

The design mission profile is illustrated in Figure 7-5. The more interesting and unusual inputs are discussed for this case while the more routine information is only listed. A complete copy of the program printout follows the description of the input.

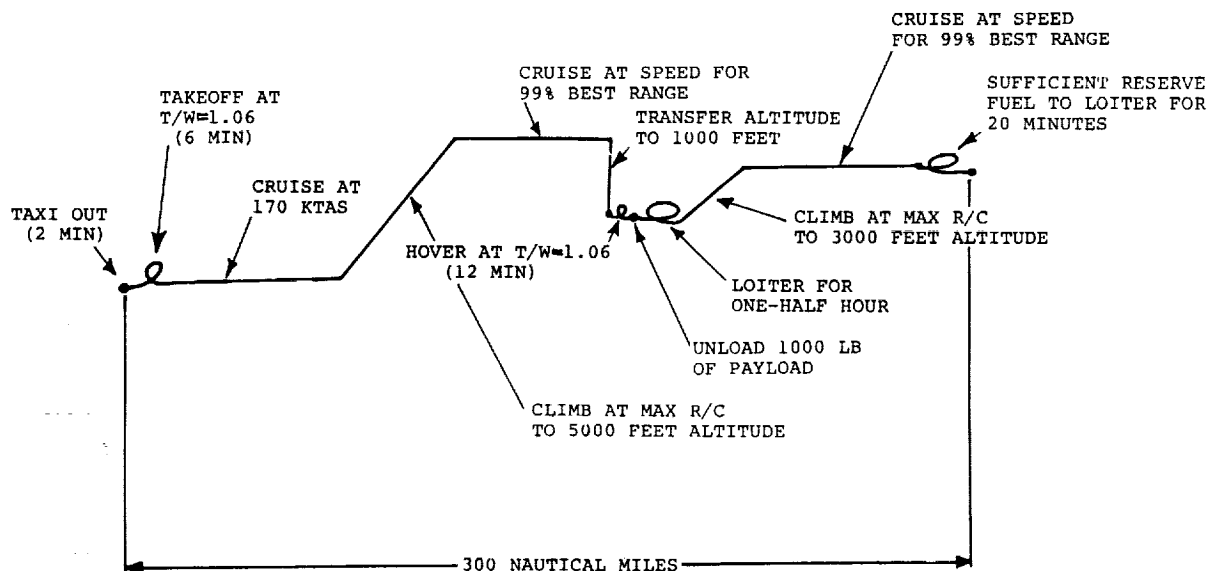


Figure 7-5. Design MSN - Sample Case Number 1.

SAMPLE CASE NO. 1

GENERAL INFORMATION SHEET

<u>VARIABLE</u>	<u>LOCATION</u>	<u>VALUE ASSIGNED</u>	<u>REMARKS</u>
OPTIND	0001	1.0	Sizing run
OPTIONAL PRINT	0002	1.0	Detailed printout de- sired
DRGIND	0003	2.0	GW/Fe Drag trend uti- lized
OSWIND	0004	0	User inputs Oswald efficiency factor
CNFIND	0005	1.0	Single-rotor helicopter
AUXIND	0006	4.0	Compound helicopter
RDMIND	0007	4.0	Main rotor diameter sized based on input disc loading; solidity sized based on input C_T/σ
FIXIND	0008	1.0	Program sizes primary engines
ROTIND	0009	1.0	Short form rotor per- formance method used
S_W IND	0010	3.0	Wing area sized by ma- neuver conditions
b_W IND	0011	1.0	Wing span sized based on input wing span/rotor diameter ratio
AIPIND	0012	2.0	Independent auxiliary engines
ENGIND	0013	0	Turboshaft auxiliary in- dependent engines
FIXINDI	0014	1.0	Program sizes auxiliary independent engines

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
TRDIND	0015	1.0	Tail rotor diameter sized based on tail rotor/main rotor diameter trend
TRSIND	0016	2.0	Tail rotor solidity sized based on input C_T/σ
VTFIND	0017	2.0	Vertical Fin area sized to meet configuration anti-torque requirements upon loss of tail rotor
HTIND	0018	2.0	Horizontal tail volume coefficient input
MRPIND	0019	0	Main rotor position (on fuselage) input by user
ESCIND	0022	2.0	Primary engines sized for either takeoff or cruise
WG ₀	0023	25000	First guess at design gross weight
h ₀	0024	0	Start altitude
R ₀	0025	0	Starting range
t ₀	0026	0	Starting time
h _{OPT} IND	0027	0	Cruise at specified altitudes
M _{MO}	0028	0.33	Maximum operating Mach number
V _{MO}	0029	220	Maximum operating EAS knots
V _{DIVE}	0030	220	Design dive speed, knots EAS

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
M _{LF}	0031	3.5	Maneuver load factor
K ₁	0032	1.0	Factor on mission fuel burned to give reserve fuel, i.e., 1.1 would give 10 percent reserves
δW_f	0033	0	Fixed fuel increment for reserves or other use
K _{FF}	0034	1.05	Increase basic engine SFC by 5 percent
SGTIND	0035	1.0	Taxi
	0036	2.0	Takeoff
	0037	4.0	Cruise
	0038	3.0	Climb
	0039	4.0	Cruise
	0040	9.0	Transfer altitude
	0041	2.0	Takeoff
	0042	8.0	Change payload
	0043	6.0	Loiter
	0044	3.0	Climb
	0045	4.0	Cruise
	0046	60.0	Loiter (reserve fuel)
	0047	100.0	End of case

Sequence of Design Mission

HELICOPTER DIMENSIONAL INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
b_w/D	0103	0.5	Wing span/main rotor diameter ratio
$(t/c)_R$	0105	0.20	Wing root thickness/chord ratio
$(t/c)_T$	0106	0.12	Wing tip thickness/chord ratio
$\Lambda_c/4$	0107	0	Quarter-chord mean sweep angle, degrees
λ	0108	0.5	Wing taper ratio (tip chord/root chord)
C_F/C	0109	1.0	Ratio of download alleviating flap chord to wing chord (1.0 signifies a fully tilting wing)
h'/h_F	0110	0.20	Ratio of vertical wing position on fuselage as a fraction of fuselage height
CL_D	0111	0.8	Wing design lift coefficient
AR_{HT}	0112	4.0	Horizontal tail aspect ratio
l_{TH}'	0113	1.15	Ratio of horizontal tail moment arm to main rotor radius
$(t/c)_{HT}$	0114	0.12	Horizontal tail thickness/chord ratio
$\bar{V}_H$	0115	0.0162	Horizontal tail volume coefficient
λ_H	0116	0.5	Horizontal tail taper ratio
$\Delta S_{wet}/S_F$	0120	0	Fuselage wetted area ratio

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
ΔS_{wet}	0121	0	Incremental fuselage wetted area
h_F	0122	7.0	Fuselage height
W_F	0123	6.5	Fuselage width
$(\ell/d)_P$	0124	1.3	Fineness ratio of nose
$(\ell/d)_T$	0125	1.0	Fineness ratio of tail
ℓ_C	0126	12.	Constant diameter section length
ℓ_{RW}	0127	0	Length of ramp well
X_M/ℓ_B	0128	0.55	Main rotor position aft of the nose as a fraction of main fuselage length
ℓ_{TB}/d_{TB}	0129	5.0	Fineness ratio of tail boom
d_{TTB}/d_{TB}	0130	0.3	Ratio of average tail boom tip diameter to average tail boom diameter
$k_{T. \text{ STING}}$	0131	1.1	Tail boom extends aft of the tail rotor disc by 10 percent of the tail rotor radius
λ_{VT}	0136	0.45	Vertical tail taper ratio
$(t/c)_{VT}$	0137	0.15	Vertical tail thickness/chord ratio
ζ_{VT}	0138	0.80	Vertical tail fin/tail rotor overlap ratio
K_Z	0139	0.85	Vertical position of the tail rotor center (relative to the vertical fin root chord) as a fraction of tail rotor radius

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
C_{LDES}	0140	0.5	Vertical tail fin design lift coefficient
V_{DES}	0141	180	Vertical tail fin sized to provide aircraft directional stability at 180 kts in event of tail rotor loss
Z_1	0142	0.035	Primary engine nacelle constants
Z_2	0143	2.0	
Z_3	0144	0.078	
Z_4	0146	0.035	Auxiliary independent engine nacelle constants
Z_5	0147	2.0	
Z_6	0148	0.078	
$(t/c)_{RF}$	0152	0.40	Forward rotor pylon root thickness/chord ratio
$(t/c)_{TF}$	0153	0.20	Forward rotor pylon tip thickness/chord ratio
AR_{FP}	0154	0.5	Forward rotor pylon aspect ratio
λ_{FP}	0155	0.4	Forward rotor pylon taper ratio
h_{p1}	0156	3.0	Forward rotor pylon height

MAIN ROTOR DIMENSIONAL DATA SHEET

ROTOR CYCLE NO.	0171	3	Rotor blade section aerodynamic characteristics selection
N_R	0172	1.0	Number of rotors
W/A	0173	11.0	Disc loading
b_{MR}	0176	4.0	Number of blades/main rotor

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
θ_{TMR}	0177	-9.0	Main rotor twist (deg)
X_{CMR}	0178	0.25	Main rotor blade cutout as a fraction of radius
X_{MR}	0179	0.075	Main rotor blade attachment point as a fraction of radius
$(t/c)_{.25R}$	0180	0.10	Rotor blade thickness/chord at 25 percent radius
V_{TIP}	0181	725	Main rotor tip speed
$(C_T/\sigma)_H$	0182	0.12	Rotor "lift coefficient" for hover sizing solidity
T/W	0183	1.06	Rotor design thrust/weight ratio
$V_{KT}(c)$	0184	165	Cruise flight conditions for sizing rotor solidity
$h_c(c)$	0185	3000	
ΔT_{INC}	0186	43.2	
$(C_T/\sigma)_{CR}$	0187	0.110	Rotor "lift coefficient" for sizing rotor solidity in cruise flight
$g_{REQM'T}$	0188	1.75	Total g requirement, helicopter must satisfy
g (ROTOR)	0189	1.35	Maneuver g's carried by main rotor
N (ROTOR LOADING)	0190	1.00	Rotor lift/GW for lg cruise flight rotor solidity sizing
V_{CEH1}	0191	1.53	Main rotor vertical rate-of-climb efficiency factors
V_{CEH2}	0192	0	
K_{PCLIMB}	0193	0.85	Helicopter forward flight climb efficiency

TAIL ROTOR DIMENSIONAL DATA SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
b_{TR}	0203	5.0	No. of blades/tail rotor
θ_{TTR}	0204	-4.0	Tail rotor twist (deg)
X_{CTR}	0205	0.3	Tail rotor blade cutout as a fraction of radius
X_{TR}	0206	0.075	Tail rotor blade attachment point as a fraction of radius
V_{TTR}	0207	690	Tail rotor tip speed
$(C_T/\sigma)_{DES} (H)$	0208	0.17	Tail rotor limiting design rotor "lift coefficient"
$\ddot{\psi}$	0209	0.30	Helicopter yaw acceleration, rad/sec ²
$\dot{\psi}$	0210	0.75	Helicopter yaw rate, rad/sec
C_{TG}/C_{TNET}	0211	1.00	Vertical tail fin/tail rotor sideload ratio (when input as 1.00, program calculates a value of C_{TG}/C_{TNET} based on tail fin/rotor geometry)
$\bar{c}$	0212	0.72	Tail rotor induced velocity ratio for a pusher type tail rotor (see fig. 4-21, sect. 4.8)
K_{ZZZ}	0213	1.00	Single rotor helicopter yaw moment of inertia trend adjustment factor (nominally = 1.00)
$g_{MR/TR}$	0214	1.0	Gap between main and tail rotor disc (ft)

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
K_{TRS}	0215	1.05	Tail rotor solidity increased 5 percent over that dictated by hovering turn requirements
C_{LFIN}	0216	0	Vertical tail fin operating cruise lift coefficient

PRIMARY ENGINE SIZING INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
PRIMARY ENGINE CYCLE NO.	0217	1.761	Primary engine selection
N_P	0219	2.0	No. of primary engines
XMSNIND	0220	2.0	Drive system rated at power required to hover or cruise (more critical of the two conditions selected by program)
SHP_{MRX}/SHP_{MR}^*	0221	1.0	Main rotor drive system is rated at 100 percent of main rotor design power
η_T	0223	0.97	Transmission efficiency
ΔSHP_{ACC}	0224	100	Accessory power losses
SHP_{TRX}/TRP^*	0225	1.0	Tail rotor drive system is rated at 100 percent of tail rotor design power
SHP_{ARX}/SHP_{AUX}^*	0226	1.0	Aux propulsion drive system is rated at 100 percent of aux propulsion design power
$h_{TO} (H)$	0227	4000	Design point hover altitude (engine sizing)

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
(T/W) _D	0228	1.06	Configuration design point hover thrust/weight ratio
(N _{II} /N _{II} MAX) _{TO}	0230	1.105	Main rotor operating at 100 percent of hover tip speed (725 fps); i.e.
			$\left(\frac{N_{II}}{N_{II}MAX}\right)_{TO} \left(\frac{N_{II}MAX}{N_{II}*}\right) V_T$ $= (1.105) (.905) (725)$ $= 725$
N _{PSD}	0231	0	No. of engines inoperative at hover design point conditions
SHP _E /SHP*	0232	0.95	Engines sized to permit operation in hover (OGE) at 95 percent of the maximum rated power
h _C	0235	3000	Design point (cruise) altitude (engine sizing)
V _C	0236	170	Design point cruise speed (engine sizing)
(T _{AUX} /T _{TOT}) _C	0239	0.75	75 percent of propulsive thrust provided by aux. propulsion at cruise conditions for engine sizing
CL _{D P}	0240	0.3	Wing operating lift coefficient at cruise conditions for engine sizing
(N _{PSD}) _C	0241	0	No. of primary engines shut down during cruise (for engine sizing)

AUXILIARY INDEPENDENT ENGINE SIZING INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
AUX PROPULSION ENGINE CYCLE NO.	0242	1.761	Auxiliary independent engine selection
N_p	0245	1.0	Helicopter has one aux. independent engine
POWIND	0246	2	Aux. independent engine sized to provide 75 percent of configuration propulsive thrust at NRP

PROPELLER DATA REQUIRED FOR COMPOUND HELICOPTER AUX PROPULSION INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
V_{TAR}	0249	900	Propeller tip speed
DIA	0250	10	Propeller diameter
X_{AR}	0251	0.075	Propeller blade attachment point as fraction of radius
η_{TAUX}	0252	0.97	Auxiliary drive system transmission efficiency
η_{pIND}	0253	0	"Point" propeller efficiencies specified for climb and cruise
η_{p3}	0254	0.82	Propeller efficiency in climb
AF/Blade	0257	140	3-way propeller, 140 activity factor/blade
No. of Blades	0258	3	

HELICOPTER AERODYNAMICS INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
(GW/Fe)	0312	1130	Drag trend constants derived from data such as illustrated by fig. 4-26, section 4-9
K_{FED}	0313	.555	
e	0314	0.75	
TFEF	0315	1.0	Tail fin aspect ratio effectiveness factor (nominally = 1.0)
$(Re/\ell)_i$	0328	1.464×10^6	Mean Reynolds No./ft based on primary engine cruise sizing flight conditions
C_{l_α}	0329	6.28	Wing 2-D lift curve slope
$C_L - C_D$ TABLE	0330 -0345	See input values on sample output sheets	

ROTOR LIMITS INFORMATION SHEET

0347 -0377	This case has been run using "dummy" rotor limit values (as explained in Sec. 7.3). For actual "dummy" values used, refer to the appropriate locations on the output sheets
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HELICOPTER WEIGHT INFORMATION

2601 -2673	This is fully explained in Section 4.11. Refer to the output sheets for the actual values used.
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TAXI INFORMATION

t_T	0411	0.0333	Taxi for 2 minutes
K_{FI}	0431	1.0	Auxiliary engine fuel flow multiplicative factor

TAKEOFF, HOVER, AND LANDING INFORMATION

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
TOLIND	0461	1.0	Specify required T/W for hover out-of-ground effect
	0462	1.0	
t _H	0551	0.1	Hover for 6 minutes
	0552	0.2	Hover for 12 minutes
CLIMB INFORMATION			
CLMIND	0571	1.0	Climb at maximum rate of climb, limited by NRP available
	0572	1.0	
T _{AUX} /T _{TOT}	0691	0	All propulsive thrust provided by main rotor
	0692	0	
C _{LWING}	0601	0.3	Wing operating C _L in climb
	0602	0.4	
N _{PSDCL}	0681	1.0	One primary engine shut down during climb
	0682	1.0	
N _{PSDiCL}	0701	1.0	Auxiliary independent engine shut down during climb
	0702	1.0	
CRUISE INFORMATION			
CRSIND	0721	2.0	Cruise at specified TAS
	0722	4.0	Cruise at 99 percent best range speed
	0723	4.0	
T _{AUX} /T _{TOT}	0841	0.55	Propeller/main rotor propulsive thrust split during cruise segments
	0842	0.60	
	0843	0.70	
C _{LWING}	0751	0.5	Wing operating C _L in cruise
	0752	0.5	
	0753	0.5	
R _{MAX}	0791	60	Values of range at end of each cruise
	0792	150	
	0793	300	

LOITER INFORMATION

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
T_{AUX}/T_{TOT}	1111	0.35	Propeller/main rotor propulsive thrust split in loiter
	1112	0.35	
C_{LW}	1041	0.4	Wing operating C_L in loiter
	1042	0.4	
t_L	1081	0.5	Loiter for 30 minutes
	1082	0.25	Loiter for 15 minutes for reserve fuel pur- poses

TRANSFER ALTITUDE SHEET

ΔW_{PL}	1161	-1000	Unload 1000 pounds of payload after 12 min- utes of hovering
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PRIMARY ENGINE CYCLE DATA; NON-STANDARD PERFORMANCE

$N2_{IND}$	1204	2.0	A non-optimum N_{II} vari- ation will be used
$N_{II_{MAX}}/N_{II}^*$	1223	0.905	Maximum power turbine speed is 90.5 percent of rated value (static, max power, sea level standard)

AUX. INDEPENDENT ENGINE CYCLE DATA; NON-STANDARD PERFORMANCE

$N2_{INDI}$	2204	2.0	A non-optimum N_{II} vari- ations will be used
$N_{II_{MAX}}/N_{II}^*$	2223	0.905	Maximum power turbine speed is 90.5 percent of rated value (static max power, sea level standard)

The sample case output follows:

H E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM B-91

THE FOLLOWING IS A CARD BY CARD REPRODUCTION OF THE INPUT DECK FOR THIS CASE

LCC. CORRESPONDS TO LOCATION NUMBER GIVEN ON INPUT SHEET
 NUM STANDS FOR THE NUMBER OF SEQUENTIAL INPUT VALUES STARTING WITH LCC. (MAX. =5)
 VAL EQUALS VALUE FOR VARIABLE CORRESPONDING TO LCC.
 VAL1 VALUE CORRESPONDING TO LCC.+0001
 VAL2 VALUE CORRESPONDING TO LCC.+0002
 ETC.

LCC.	NUM	VAL	VAL1	VAL2	VAL3	VAL4
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NOTE : IN USING AUXILIARY ENGINES ; AUXILIARY ENGINE CYCLE INPUT LOCATIONS CAN BE CREATED
 BY PLACING A 66666 CARD IN FRONT AND BEHIND A STANDARD ENGINE CYCLE

1601	2	3.0000	-9.0000			
1613	5	0.99500E-02	-0.28000E-01	0.26200	0.27600	2.4500
1608	1	0.86500				
1609	5	0.10500E-01	2.8200	0.90000E-01	1.1700	0.12400E-02
1614	2	0.75800	0.74300			
1616	5	10.000	0.0	0.40000E-02	0.70000E-02	0.90000E-02
1621	5	0.10000E-01	0.11000E-01	0.11500E-01	0.12000E-01	0.15500E-01
1626	5	0.22000E-01	1.0180	1.0850	1.1540	1.2330
1631	5	1.2790	1.3140	1.3270	1.3370	1.3640
1636	1	1.3970				
1	5	1.0000	1.0000	2.0000	0.0	1.0000
6	5	4.0000	4.0000	1.0000	1.0000	3.0000
11	5	1.0000	2.0000	0.0	1.0000	1.0000
16	4	2.0000	2.0000	2.0000	0.0	
22	4	2.0000	25000.	0.0	0.0	
26	5	0.0	0.0	0.33000	220.00	220.00
31	5	3.5000	1.0000	0.0	1.0500	1.0000
36	5	2.0000	4.0000	3.0000	4.0000	9.0000
41	5	2.0000	8.0000	6.0000	3.0000	4.0000
46	2	60.000	100.00			
103	1	0.50000				
105	5	0.20000	0.12000	0.0	0.50000	1.0000
110	2	0.20000	0.80000			
112	5	4.0000	1.1500	0.12000	0.16200E-01	0.50000
120	2	0.0	0.0			
122	5	7.0000	6.5000	1.3000	1.0000	12.000
127	5	0.0	0.55000	5.0000	0.30000	1.1000
136	5	0.45000	0.15000	0.80000	0.85000	0.50000
141	1	180.00				
142	3	0.35000E-01	2.0000	0.78000E-01		
146	3	0.35000E-01	2.0000	0.78000E-01		
150	2	0.0	0.0			
152	5	0.40000	0.20000	0.50000	0.40000	3.0000
171	1	3.0000				
172	2	1.0000	11.000			
176	5	4.0000	-9.0000	0.25000	0.75000E-01	0.10000E-01
181	1	725.00				
182	2	0.12000	1.0600			

184	5	165.00	3000.0	43.200	0.11000	1.7500
189	2	1.3500	1.0000			
191	2	1.5300	0.0			
193	1	0.85000				
203	5	5.0000	-4.0000	0.30000	0.75000E-01	690.00
208	5	0.17000	0.30000	0.75000	1.0000	0.72000
213	3	1.0000	1.0000	1.3500		
216	1	0.0				
217	1	1.7610				
219	1	2.0000				
220	2	2.0000	1.0000			
223	2	0.97000	1.00.00			
225	2	1.0000	1.0000			
227	5	4000.0	1.0000	50.300	1.1050	0.0
232	2	0.95000	0.0			
234	5	2.0000	3000.0	170.00	43.200	1.1050
239	3	0.75000	0.30000	0.0		
242	1	1.7610				
245	1	1.0000				
246	2	2.0000	1.1050			
248	5	1.0000	900.00	10.000	0.75000E-01	0.97000
253	1	0.0				
254	2	0.82000	0.80000			
257	2	140.00	3.0000			
261	1	4.0000				
262	4	0.0	0.20000	0.40000	0.80000	
272	4	0.85000	0.83000	0.80000	0.78000	
312	4	1130.0	0.55500	0.75000	1.0000	
323	1	0.0				
327	1	1.0200				
328	1	0.14640E-07				
329	1	6.2800				
330	1	7.0000				
331	5	0.0	0.20000	0.40000	0.60000	0.80000
336	2	1.0000	1.4000			
339	5	0.60000E-02	0.62000E-02	0.70000E-02	0.80000E-02	0.95000E-02
344	2	0.12000E-01	0.20000E-01			
347	2	3.0000	3.0000			
349	3	0.0	0.50000	1.0000		
354	3	0.0	0.50000	1.0000		
361	3	1.0000	1.0000	1.0000		
368	3	1.0000	1.0000	1.0000		
375	3	1.0000	1.0000	1.0000		
2602	3	2200.0	450.00	2000.0		
2605	3	100.00	0.0	0.0		
2608	5	0.0	0.0	0.0	0.0	0.0
2613	5	25.000	25.000	25.000	0.0	0.0
2618	4	30.000	0.18000	25.000	0.0	
2622	5	125.00	2.0800	0.40000E-01	0.80000	0.0
2627	5	1.0000	0.0	2.0600	2.0000	24.100
2632	5	1.0000	400.00	0.0	175.00	0.0
2637	5	44.000	2.2000	61.000	0.28000	1.1500
2642	5	14.200	14.200	1.0000	1.0000	250.00
2647	5	3.0000	250.00	250.00	1.0000	0.11000
2652	2	0.17000	0.17000			
2654	5	1.0000	1.0000	1.0000	1.0000	1.0000
2659	5	1.0000	1.0000	1.0000	1.0000	1.0000
2664	1	1.0000				
2665	5	1.0000	1.0000	1.0000	1.0000	1.0000
2670	4	1.0000	1.0000	1.0000	1.0000	

401	1	0.0			
411	1	0.3300E-01			
421	1	0.0			
431	1	1.0000			
441	1	1.1050			
461	2	1.0000	1.0000		
481	2	0.0	0.0		
501	2	0.0	0.0		
511	2	0.0	0.0		
521	2	1.0000	1.0000		
531	2	0.2000E-01	0.2000E-01		
541	2	1.1050	1.1050		
551	2	0.1000E-00	0.20000		
571	2	1.0000	1.0000		
591	2	0.0	0.0		
601	2	0.30000	0.40000		
611	2	0.0	0.0		
621	2	500.00	500.00		
631	2	2.0000	2.0000		
641	2	5000.0	3000.0		
651	2	1.1050	1.1050		
661	2	6.0000	6.0000		
671	2	1.1050	1.1050		
681	2	1.0000	1.0000		
691	2	0.0	0.0		
701	2	1.0000	1.0000		
721	3	2.0000	4.0000	4.0000	
731	1	170.00			
741	3	0.0	0.0	0.0	
751	3	0.50000	0.50000	0.50000	
761	3	0.0	0.0	0.0	
771	3	15.000	15.000	15.000	
781	3	2.0000	2.0000	2.0000	
791	3	60.000	150.00	300.00	
801	3	1.1050	1.1050	1.1050	
811	3	0.0	0.0	0.0	
821	3	1.1050	1.1050	1.1050	
831	3	0.0	0.0	0.0	
841	3	0.55000	0.60000	0.70000	
851	3	0.0	0.0	0.0	
1031	2	0.0	0.0		
1041	2	0.40000	0.40000		
1051	2	0.0	0.0		
1061	2	0.5000E-01	0.5000E-01		
1071	2	1.1050	1.1050		
1081	2	0.50000	0.25000		
1091	2	1.1050	1.1050		
1101	2	0.0	0.0		
1111	2	0.35000	0.35000		
1121	2	0.0	0.0		
1131	2	0.0	0.0		
1161	1	-1000.0			
1171	1	0.1000E-01			
1181	1	1000.0			
1191	1	1.0000			
1201	5	0.0	0.0	0.0	2.0000
1206	1	0.0			
1223	1	0.90500			
2201	5	0.0	0.0	0.0	2.0000
2206	1	0.0			

2223	1	0.90500					
1201	5	0.0	0.0	0.0	2.0000	0.0	
1206	1	0.0					
1223	1	0.90500					
1301	5	1.7610	0.15900	0.0	0.32000E-01	950.00	
1306	5	1100.0	1856.0	2000.0	2000.0	8.0000	
1311	5	950.00	1200.0	1400.0	1600.0	1800.0	
1316	5	2000.0	2200.0	2600.0	5.0000	0.0	
1321	4	0.20000	0.40000	0.60000	0.80000		
1326	5	0.25000E-01	0.25700E-01	0.27800E-01	0.31300E-01	0.36200E-01	
1332	5	0.16300	0.16760	0.18130	0.20410	0.23600	
1338	5	0.33500	0.34440	0.37250	0.41940	0.48510	
1344	5	0.54400	0.55920	0.60490	0.68110	0.78770	
1350	5	0.77000	0.79160	0.85620	0.96400	1.1150	
1356	5	1.0000	1.0280	1.1120	1.2520	1.4480	
1362	5	1.2000	1.2336	1.3344	1.5024	1.7376	
1368	5	1.5500	1.5934	1.7236	1.9406	2.2444	
1374	5	8.0000	950.00	1200.0	1400.0	1600.0	
1379	5	1800.0	2000.0	2200.0	2600.0	5.0000	
1384	5	0.0	0.20000	0.40000	0.60000	0.80000	
1390	5	0.65000E-01	0.65100E-01	0.65300E-01	0.67000E-01	0.71000E-01	
1396	5	0.11500	0.11630	0.11800	0.12800	0.14000	
1402	5	0.18000	0.18100	0.19000	0.20800	0.22700	
1408	5	0.26000	0.26130	0.27300	0.29500	0.32500	
1414	5	0.34200	0.34700	0.36200	0.38900	0.42500	
1420	5	0.42500	0.43500	0.45100	0.48600	0.51700	
1426	5	0.50000	0.51100	0.53000	0.56000	0.61000	
1432	5	0.62600	0.63130	0.66000	0.71800	0.78000	
1438	4	3.0000	950.00	1600.0	2600.0		
1447	4	3.0000	0.0	0.40000	0.80000		
1454	3	0.26000	0.27100	0.29000			
1460	3	0.82000	0.84000	0.90000			
1466	3	1.0900	1.1180	1.1650			
1502	5	8.0000	950.00	1200.0	1400.0	1600.0	
1507	5	1800.0	2000.0	2200.0	2600.0	5.0000	
1512	5	0.0	0.20000	0.40000	0.60000	0.80000	
1518	5	0.26000	0.26500	0.27100	0.28000	0.29000	
1524	5	0.52000	0.52700	0.54000	0.56000	0.59000	
1530	5	0.68000	0.69000	0.70500	0.73000	0.76000	
1536	5	0.82000	0.82400	0.84000	0.86800	0.90000	
1542	5	0.92000	0.93000	0.95000	0.98000	1.0200	
1548	5	1.0000	1.0020	1.0200	1.0500	1.0900	
1554	5	1.0520	1.0550	1.0700	1.1000	1.1310	
1560	5	1.0900	1.1000	1.1180	1.1350	1.1650	
2201	5	0.0	0.0	0.0	2.0000	0.0	
2206	1	0.0					
2223	1	0.90500					
2301	5	1.7610	0.15900	0.0	0.32000E-01	950.00	
2306	5	1100.0	1856.0	2000.0	2000.0	8.0000	
2311	5	950.00	1200.0	1400.0	1600.0	1800.0	
2316	5	2000.0	2200.0	2600.0	5.0000	0.0	
2321	4	0.20000	0.40000	0.60000	0.80000		
2326	5	0.25000E-01	0.25700E-01	0.27800E-01	0.31300E-01	0.36200E-01	
2332	5	0.16300	0.16760	0.18130	0.20410	0.23600	
2338	5	0.33500	0.34440	0.37250	0.41940	0.48510	
2344	5	0.54400	0.55920	0.60490	0.68110	0.78770	
2350	5	0.77000	0.79160	0.85620	0.96400	1.1150	
2356	5	1.0000	1.0280	1.1120	1.2520	1.4480	
2362	5	1.2000	1.2336	1.3344	1.5024	1.7376	
2368	5	1.5500	1.5934	1.7236	1.9406	2.2444	

2374	5	8.0000	950.00	1200.0	1400.0	1600.0
2379	5	1800.0	2000.0	2200.0	2600.0	5.0000
2384	5	0.0	0.20000	0.40000	0.60000	0.80000
2390	5	0.05000E-01	0.65100E-01	0.65300E-01	0.67000E-01	0.71000E-01
2390	5	0.11500	0.11600	0.11800	0.12800	0.14000
2402	5	0.18000	0.18100	0.19000	0.20800	0.22700
2409	5	0.26000	0.26100	0.27300	0.29500	0.32500
2414	5	0.34200	0.34700	0.36200	0.38900	0.42500
2420	5	0.42500	0.43500	0.45100	0.48600	0.51700
2426	5	0.50000	0.51100	0.53000	0.56000	0.61000
2432	5	0.62000	0.63100	0.66000	0.71800	0.78000
2438	4	0.0000	950.00	1600.0	2600.0	
2447	4	0.0000	0.0	0.40000	0.80000	
2454	3	0.26000	0.27100	0.29000		
2460	3	0.82000	0.84000	0.90000		
2466	3	1.09000	1.11800	1.1650		
2502	5	8.0000	950.00	1200.0	1400.0	1600.0
2507	5	1800.0	2000.0	2200.0	2600.0	5.0000
2512	5	0.0	0.20000	0.40000	0.60000	0.80000
2518	5	0.26000	0.26500	0.27100	0.28000	0.29000
2524	5	0.52000	0.52700	0.54000	0.56000	0.59000
2530	5	0.68000	0.69000	0.70500	0.73000	0.76000
2530	5	0.82000	0.82400	0.84000	0.86800	0.90000
2542	5	0.92000	0.93000	0.95000	0.98000	1.0200
2548	5	1.0000	1.0020	1.0200	1.0500	1.0900
2554	5	1.0520	1.0550	1.0700	1.1000	1.1310
2560	5	1.0900	1.1000	1.1180	1.1350	1.1650

WJ = 1.250000E 05 WFA = 0.251484E 00 WFR = 0.251190E 52
 WJ = 0.250000E 05 WFA = 0.712128E 04 WFR = 0.491322E 04
 WJ = 0.218456E 05 WFA = 0.539664E 04 WFR = 0.436166E 04
 WJ = 0.190626E 05 WFA = 0.402145E 04 WFR = 0.387086E 04

H E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM B-91

SINGLE ROTOR COMPOUND HELICOPTER AUX. INDEPENDENT T/SHAFT CRUISE PROPULSION

S I Z E D A T A THIS RUN CONVERGED IN 4 ITERATIONS

GROSS WEIGHT = 18589. LB

FUSELAGE

LF	LENGTH(BODY+TAILBOOM)	51.0 FT.
LC	LENGTH(CABIN)	12.0 FT.
LB	LENGTH(BODY)	27.5 FT.
LTR	LENGTH(TAILBOOM)	23.5 FT.
XM	FWD. ROTOR LOCATION	15.1 FT.
WF	WIDTH	6.5 FT.
SF	WETTED AREA	755.6 SQ. FT.

WING

AP	ASPECT RATIO	4.51
SW	AREA	119.4 SQ. FT.
9W	SPAN	23.2 FT.
CHARW	MEAN CHORD	5.1 FT.
LAMBDA C/4	QUARTER CHORD SWEEP	0.0 DEG.
LAMBDA	TAPER RATIO	0.500
(T/C)R	ROOT THICKNESS/CHORD	0.200
(T/C)T	TIP THICKNESS/CHORD	0.120
WG/SW	WING LOADING	155.7 LBS/SQ. FT.
GDW	ROTOR/WING GAP	8.6 FT.
CF/C	FLAP CHORD/MEAN CHORD RATIO	1.000

HJR. TAIL

AFHT	ASPECT RATIO	4.000
SHT	AREA	37.4 SQ. FT.
9HT	SPAN	12.2 FT.
CHARHT	MEAN CHORD	3.1 FT.
LAMBDA H	TAPER RATIO	0.500
(T/C)HT	THICKNESS/CHORD	0.120
LTH	HCR. TAIL ARM	26.7 FT.

VERT. TAIL

ARVT	ASPECT RATIO	1.649
SVT	AREA	20.6 SQ. FT.
9VT	SPAN	5.8 FT.
CHARVT	MEAN CHORD	3.5 FT.
LAMBDA VT	TAPER RATIO	0.450
ZTR	TAIL ROTOR(VERT.) LOCATION	4.7 FT.
ZFTAVT	TAIL ROTOR/VERT. TAIL OVERLAP RATIO	0.800

(T/C)VT	THICKNESS/CHORD	0.150
MAIN ROTOR PYLON		
AR	ASPECT RATIO	0.500
SFP	WETTED AREA	39.1 SQ. FT.
FAPF	FRONTAL AREA	6.2 SQ. FT.
HPI	HEIGHT	3.3 FT.
CBARFP	MEAN CHORD	6.0 FT.
LAMPDA FP	TAPER RATIO	0.400
(T/C)F	ROOT THICKNESS/CHORD	0.400
(T/C)T	TIP THICKNESS/CHORD	0.200
PRIMARY ENGINE NACELLE		
LN	LENGTH	5.8 FT.
DN	MEAN DIAMETER	2. FT.
SN	WETTED AREA(TOTAL FOR ALL ENGINES)	62.6 SQ. FT.
AUXILIARY INDEPENDENT ENGINE NACELLE		
LNI	LENGTH	4.6 FT.
ONI	MEAN DIAMETER	1. FT.
SNI	WETTED AREA(TOTAL FOR ALL ENGINES)	16. SQ. FT.
AUXILIARY INDEPENDENT ENGINE NACELLE STRUT		
SSTW	WETTED AREA(TOTAL)	0. SQ. FT.
RNS	SPAN	0.0 FT.
CNS	MEAN CHORD	2.6 FT.
PROPELLER(AUXILIARY PROPULSION)		
DAP	DIAMETER	10.0 FT.
AF	ACTIVITY FACTOR PER BLADE	140.0
SIGAR	SOLIDITY	0.171
NPA	NO. OF PROPELLERS	1.
NO. BLADES	NO. OF BLADES/PROP	3.
VTIP	TIP SPEED	930. FT./SEC
MAIN ROTOR		
DMP	DIAMETER	46.4 FT.
SIGMR	SOLIDITY	0.128
WG/A	DISC LOADING	11.0 LB./SQ. FT.
CT/SIGMA	THRUST COEFF./SOLIDITY	0.110
NP	NO. OF ROTORS	1.
NO. BLADES	NO. OF BLADES/ROTOR	4.
THETA	BLADE TWIST	-9.000 DEG.
XC	BLADE CUTOUT/RADIUS RATIO	0.250
VTIP	TIP SPEED	725. FT./SEC.
TAIL ROTOR		
DTP	DIAMETER	11.1 FT.
SIGTR	SOLIDITY	0.313
(T/A)NET	NET DISC LOADING	17.4 LB./SQ. FT.
CT/SIGMA	THRUST COEFF./SOLIDITY	0.170
NO. BLADES	NO. OF BLADES/ROTOR	5.
THETA	BLADE TWIST	-4.000 DEG.
XCTR	BLADE CUTOUT/RADIUS RATIO	0.300
G	MAIN/TAIL ROTOR GAP	1.0 FT.
VTIP	TIP SPEED	690. FT./SEC.

H E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM 8-91

WEIGHTS DATA IN LBS

MLF	MANEUVER LOAD FACTOR	3.500	
GLF	GUST LOAD FACTOR	1.508	
ULF	ULTIMATE LOAD FACTOR	5.250	
PROPULSION GROUP			
	TOTAL MAIN ROTOR GROUP	1876.	
K12 WPRB	MAIN ROTOR BLADE (PER ROTOR)	1096.	
K13 WPH	MAIN ROTOR HUB (PER ROTOR)	535.	
WBF	BLADE FOLDING (PER ROTOR)	245.	
K15 WAP	AUXILIARY PROPULSION ROTOR GROUP	133.	
WCS	DRIVE SYSTEM	1725.	
K16 WPOS	MAIN ROTOR DRIVE SYSTEM	1422.	
K20 WTRDS	TAIL ROTOR DRIVE SYSTEM	121.	
K17 WACS	AUXILIARY PROPULSION DRIVE SYSTEM	182.	
K18 WEP	PRIMARY ENGINES	762.	
K19 WEA	AUXILIARY ENGINES	171.	
WPEI	PRIMARY ENGINE INSTALLATION	130.	
WAFI	AUXILIARY ENGINE INSTALLATION	29.	
WFS	FUEL SYSTEM	416.	
DELTA WP	PROPULSION GROUP WEIGHT INCREMENT	0.	
WP	TOTAL PROPULSION GROUP WEIGHT		5242.
STRUCTURES GROUP			
K8 WW	WING	746.	
WTG	TAIL GROUP	228.	
K9 WHT	HOR. TAIL	75.	
K14 WTR	TAIL ROTOR	154.	
K6 WB	FUSELAGE	1962.	
K7 WLG	LANDING GEAR	744.	
WNG	NOSE GEAR	145.	
WMG	MAIN GEAR	595.	
WTES	TOTAL ENGINE SECTION	815.	
WPES	PRIMARY ENGINE SECTION	610.	
WAES	AUXILIARY ENGINE SECTION	205.	
DELTA WST	STRUCTURE WEIGHT INCREMENT	0.	
WST	TOTAL STRUCTURE WEIGHT		3995.
FLIGHT CONTROLS GROUP			
WPFC	PRIMARY FLIGHT CONTROLS	760.	
WCC	COCKPIT CONTROLS	83.	
K1 WRC	MAIN ROTOR CONTROLS	386.	
K2 WSC	MAIN ROTOR SYSTEMS CONTROLS	261.	
K3 WFW	FIXED WING CONTROLS	0.	
WTM	TYLT MECHANISM	0.	
WSAS	SAS	30.	
WAFS	AUXILIARY FLIGHT CONTROLS	56.	
K4 WRCA	AUX. PROPULSION ROTOR CONTROLS	0.	
K5 WSCA	AUX. PROPULSION ROTOR SYS. CONTROLS	32.	
WMC	MISCELLANEOUS CONTROLS	0.	
DELTA WFC	CONTROL WEIGHT INCREMENT	100.	
WFC	TOTAL CONTROL WEIGHT		915.
WFE	WEIGHT OF FIXED EQUIPMENT		2200.
WE	WEIGHT EMPTY		12353.
WFUL	FIXED USEFUL LOAD		450.
OWE	OPERATING WEIGHT EMPTY		12803.
WPL	PAYLOAD		2000.
(WF)A	FUEL		3786.
WG	GROSS WEIGHT		18589.

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R O T O R D A T A

ROTOR CYCLE NO. 3.0000

MAIN ROTOR SOLIDITY SIZED BY MANUEVER CONDITIONS

H = 3000.0 FT., TEMP = 91.5 DEG., V = 165.0 KT.
ROTOR MANUEVER G'S = 1.350, CT/SIGMA = 0.110

TAIL ROTOR SIZED AT 1.050 TIMES THE SOLIDITY
REQUIRED TO SATISFY HOVERING TURN REQUIREMENTS AT :

H	=	4000.0 FT.
TEMP	=	95.035 DEG., F.
CTG/CTNET	=	1.221
YAW RATE	=	0.750 RAD/SEC.
YAW ACCELERATION	=	0.300 RAD/SEC ²
TAIL ROTOR POLAR		
MOM. OF INERTIA (PER BLADE)	=	9.870 SLUG/FT ²
HELICOPTER YAW		
MOM. OF INERTIA	=	39312.9 SLUG/FT ²

SAMPLE CASE NO. 1

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H E S C O M P
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P R O P U L S I O N D A T A

PRIMARY PROPULSION CYCLE NO. 1.761
TURBOSHAFT ENGINE

2. ENGINES

BHP*P MAX. STANDARD S.L. STATIC H.P. 4794. H.P.

ENGINE SIZED FOR TAKEOFF AT T/W = 1.06
H = 4000. FT, TEMPERATURE = 95.04 DEG.F.
AND 0.0 ENGINES INOPERATIVE.

AUX. INDEPENDENT PROPULSION CYCLE NO. 1.761
TURBOSHAFT ENGINE

1. ENGINES

BHP*PI MAX. STANDARD S.L. STATIC H.P. 1073. H.P.

ENGINE SIZED FOR CRUISE AT VC = 170. KNOTS,
HC = 3000. FT, TEMPERATURE = 91.50 DEG.F.
AND 0.0 ENGINES INOPERATIVE.

MAIN AND TAIL ROTOR DRIVE SYSTEM RATING 3473. H.P.

MAIN ROTOR DRIVE SYSTEM RATING 3039. H.P.

XMSN SIZED AT 100. PERCENT OF MAIN ROTOR HOVER POWER REQUIRED
AT H = 4000. FT, TEMP = 95.04 DEG.F.

TAIL ROTOR DRIVE SYSTEM RATING 434. H.P.

XMSN SIZED AT 100. PERCENT OF TAIL ROTOR HOVER POWER REQUIRED
AT H = 4000. FT, TEMP = 95.04 DEG.F.

AUXILIARY INDEPENDENT PROPULSION DRIVE SYSTEM RATING 880. H.P.

XMSN SIZED AT 100. PERCENT OF AUX. PROPULSION CRUISE POWER REQUIRED AT VC = 170. KT,
HC = 3000. FT, TEMP = 91.50 DEG.F.

SAMPLE CASE NO. 1

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A E R O D Y N A M I C S D A T A			
FE	TOTAL EFFECTIVE FLATPLATE AREA	19.534	SQFT
SWET	TOTAL WETTED AREA	1177.	SQFT
CPARF	MEAN SKIN FRICTION COEFF.	0.016697	
D W A G B R E A K D O W N I N S Q F T			
FEW	WING FE	0.765	
FFF	FUSELAGE FE	18.769	
FEFF	FORWARD(MAIN) ROTOR PYLON FE	0.0	
FEAP	AFT ROTOR PYLON FE	0.0	
FEMPH	MAIN ROTOR HUB(S) FE	0.0	
FETRH	TAIL ROTOR HUB FE	0.0	
FFVT	VERTICAL TAIL FE	0.0	
FEHT	HORIZONTAL TAIL FE	0.0	
FEN	PRIMARY ENGINE NACELLE FE	0.0	
FFNI	AUX. INDEPENDENT CRUISE ENG. NAC. FE	0.0	
FENS	AUX. INDEPENDENT CRUISE ENG. STRUT FE	0.0	
DELTA FE	INCREMENTAL FE	0.0	
A E R O D Y N A M I C C O E F F .			
AS		18.76888	
AC		1.06804	
A7		0.09420	
A8		0.00011	
A9		0.22017	
E	WING LIFT EFFICIENCY FACTOR	0.75000	
EVT	VERTICAL TAIL LIFT EFFICIENCY FACTOR	0.87677	

H E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM 8-91

MISSION PERFORMANCE DATA

TAXI FOR 0.033 HRS. AT GROUND IDLE ENGINE RATING

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRESS. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PERF	TOTAL FUEL FLOW (LBS/Hr)	AUX. TURB. TEMP. (R)	AUX. ENG. CODE	AUX. ENG. PERF	AUX. ENG. FUEL FLOW (LBS/Hr)
0.0	0.0	0.0	0.0	0.0	0.0	950.0	T	0.0	400.	950.0	T	0.0	73.
0.033	0.0	13.3	18575.	0.	0.0	950.0	T	0.0	400.	950.0	T	0.0	73.

TAKEOFF, HOVER, OR LAND AT T/M = 1.060 FOR 0.100 HRS.

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRESS. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PERF	TOTAL FUEL FLOW (LBS/Hr)	THRUST TO WEIGHT	FM	BHP	CT	CT/SIGMA
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M. ROTOR VTIP (FPS)	M. ROTOR RHP	T. ROTOR RHP	VRC RHP	PRIM. ENG. FUEL FLOW (LBS/Hr)	ROT LIM CODE
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TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRESS. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PERF	TOTAL FUEL FLOW (LBS/Hr)	DELOCN	FM	CPRO	CPIND	CDO
0.033	0.0	13.3	18575.	0.	0.0	1688.1	P	0.628	1563.	1.060	0.708	3010.	0.0093	0.073
725.0	2403.	690.0	320.	0.	1490.	73.	A		0.0	0.0	0.708	0.00011	0.00079	0.0093
0.033	0.0	44.6	18544.	0.	0.0	1686.8	P	0.626	1560.	1.060	0.708	3001.	0.0093	0.073
725.0	2496.	690.0	318.	0.	1487.	73.	A		0.0	0.0	0.708	0.00011	0.00079	0.0093
0.073	0.0	75.8	18513.	0.	0.0	1685.4	P	0.624	1558.	1.060	0.708	2993.	0.0093	0.073
725.0	2489.	690.0	317.	0.	1484.	73.	A		0.0	0.0	0.708	0.00011	0.00078	0.0093
0.093	0.0	107.0	18482.	0.	0.0	1684.0	P	0.623	1555.	1.060	0.708	2985.	0.0093	0.072
725.0	2462.	691.0	316.	0.	1481.	73.	A		0.0	0.0	0.708	0.00011	0.00078	0.0093
0.113	0.0	138.1	18451.	0.	0.0	1682.6	P	0.621	1552.	1.060	0.709	2977.	0.0093	0.072
725.0	2476.	691.0	315.	0.	1479.	73.	A		0.0	0.0	0.709	0.00011	0.00078	0.0093

CRUISE AT 170.0 KNOTS TAS, LIMITED BY NORMAL ENGINE RATING

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRESS. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PERF	FAS (KTS)	MU	CT PRIME OVER SIGMA	ALPHA D/L (DEG)	SPEC. RANGE (NMPP)	BHP
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M. ROTOR VTIP (FPS)	M. ROTOR RHP	T. ROTOR RHP	VRC RHP	PRIM. ENG. FUEL FLOW (LBS/Hr)	ETAP PROP	AUX. ENG. FUEL FLOW (LBS/Hr)	AUX. TURB. TEMP. (R)	AUX. ENG. CODE	AUX. ENG. PERF
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1.301	150.00	1775.9	16813.	1000.	0.0	1629.6	P	0.560	1410.	1.060	0.712	2618.	0.0087	0.068
725.0	2176.	690.0	267.	0.	1339.	71.	A			0.0	0.712	0.00011	0.00069	0.0093
1.321	150.00	1804.1	16785.	1000.	0.0	1628.5	P	0.559	1407.	1.060	0.712	2611.	0.0087	0.068
725.0	2170.	690.0	266.	0.	1336.	71.	A			0.0	0.712	0.00011	0.00069	0.0093

CHANGE PAYLOAD, REMOVE 1000. LB.

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PEHF	EAS (KTS)	MU	CT PRIME JVER SIGMA	ALPHA D/L (DEG)	TOTAL FUEL FLOW (LBS/HR)	BHP
1.321	150.00	1804.1	16785.	1000.	0.0	1628.5	P	0.559	1407.	1.060	0.712	2611.	0.0087	0.068
1.331	150.00	1804.1	15785.	1000.	0.0	1628.5	P	0.559	1407.	1.060	0.712	2611.	0.0087	0.068

LOITER FOR 0.500 MRS.

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PEHF	EAS (KTS)	MU	CT PRIME JVER SIGMA	ALPHA D/L (DEG)	TOTAL FUEL FLOW (LBS/HR)	BHP
1.331	150.00	1804.1	15785.	1000.	77.6	1377.0	P	0.243	76.5	0.181	0.057	-2.0	979.	1146.
725.0	957.	690.0	57.	*****	846.	0.	0.838	0.350	133.	1219.3	P	0.057	0.017	60.
0.000150	0.000160	0.000034	0.000010	0.01087	0.0	0.00044	0.000186	A	*****	*****	*****	0.400	0.007	0.940
1.381	150.00	1853.0	15736.	1000.	77.6	1376.4	P	0.242	76.5	0.181	0.056	-2.0	978.	1142.
725.0	954.	690.0	57.	*****	845.	0.	0.838	0.350	133.	1219.2	P	0.057	0.007	60.
0.000150	0.000159	0.000034	0.000010	0.01087	0.0	0.00044	0.000186	A	*****	*****	*****	0.400	0.007	0.940
1.431	150.00	1902.0	15687.	1000.	77.6	1375.8	P	0.241	76.5	0.181	0.056	-2.0	977.	1139.
725.0	951.	690.0	57.	*****	844.	0.	0.838	0.350	133.	1219.2	P	0.057	0.007	60.
0.000150	0.000158	0.000034	0.000010	0.01086	0.0	0.00044	0.000186	A	*****	*****	*****	0.400	0.007	0.940
1.481	150.00	1950.8	15638.	1000.	77.6	1375.2	P	0.241	76.5	0.181	0.056	-2.0	976.	1136.
725.0	948.	690.0	57.	*****	843.	0.	0.838	0.350	133.	1219.2	P	0.057	0.007	60.
0.000150	0.000157	0.000034	0.000010	0.01086	0.0	0.00044	0.000186	A	*****	*****	*****	0.400	0.007	0.939
1.531	150.00	1999.6	15589.	1000.	77.6	1374.7	P	0.240	76.5	0.181	0.056	-2.0	975.	1133.
725.0	944.	690.0	57.	*****	842.	0.	0.838	0.350	133.	1219.1	P	0.057	0.007	60.
0.000150	0.000156	0.000034	0.000010	0.01086	0.0	0.00043	0.000185	A	*****	*****	*****	0.400	0.007	0.939
1.581	150.00	2048.4	15540.	1000.	76.6	1374.3	P	0.240	75.5	0.178	0.056	-2.0	974.	1130.
725.0	942.	690.0	57.	*****	842.	0.	0.838	0.350	133.	1217.6	P	0.055	0.007	58.
0.000149	0.000157	0.000033	0.000009	0.01083	0.0	0.00041	0.000182	A	*****	*****	*****	0.400	0.007	0.941
1.631	150.00	2097.1	15492.	1000.	76.6	1373.8	P	0.239	75.5	0.178	0.056	-2.0	973.	1127.
725.0	939.	690.0	57.	*****	841.	0.	0.838	0.350	133.	1217.6	P	0.055	0.007	58.
0.000149	0.000156	0.000033	0.000009	0.01083	0.0	0.00040	0.000182	A	*****	*****	*****	0.400	0.007	0.940

1.681	150.00	2145.7	15443.	1001.	76.6	1373.2	P	0.238	75.5	0.178	0.055	-2.1	972.	112.
725.0	936.	690.0	57.	0.01083	840.	0.00000	0.01081	A	133.	1217.5	0.055	0.055	0.007	0.940
0.000149	0.000155	0.000033	0.000009	0.01083	0.0	0.00000	0.000181	A	0.055	0.055	0.055	0.055	0.007	0.940
1.731	150.00	2154.3	15394.	1000.	76.6	1372.6	P	0.238	75.5	0.178	0.055	-2.1	971.	112.
725.0	933.	690.0	57.	0.01083	839.	0.00000	0.01081	A	133.	1217.5	0.055	0.055	0.007	0.940
0.000149	0.000154	0.000033	0.000009	0.01083	0.0	0.00000	0.000181	A	0.055	0.055	0.055	0.055	0.007	0.940
1.781	150.00	2242.9	15346.	1000.	76.6	1372.1	P	0.237	75.5	0.178	0.055	-2.1	970.	111.
725.0	930.	690.0	57.	0.01082	838.	0.00000	0.01081	A	133.	1217.4	0.055	0.055	0.007	0.940
0.000149	0.000153	0.000033	0.000009	0.01082	0.0	0.00000	0.000181	A	0.055	0.055	0.055	0.055	0.007	0.940
1.831	150.00	2291.4	15297.	1000.	76.6	1371.5	P	0.236	75.5	0.178	0.055	-2.1	969.	111.
725.0	927.	690.0	57.	0.01082	837.	0.00000	0.01081	A	133.	1217.4	0.055	0.055	0.007	0.940
0.000149	0.000152	0.000033	0.000009	0.01082	0.0	0.00000	0.000181	A	0.055	0.055	0.055	0.055	0.007	0.940

CLIMB TO 3000. FT. WITH MAXIMUM K/C AT NORMAL ENGINE RATING

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PEHF	FAS (KTS)	MU	CT JVER	CT SIGMA	ALPHA D/L (DEG)	GAMMA (DEG)	R/C (FPM)
M. ROTOR VTIP (FPS)	M. ROTOR RHP	T. ROTOR VTIP (FPS)	T. ROTOR RHP	PROP VTIP (FPS)	PRIM. ENG. FUEL FLOW (LBS/HR)	PRIM. ENG. FUEL FLOW (LBS/HR)	ETAP PROP	AUX. ENG. FUEL FLOW (LBS/HR)	AUX. TURB. TEMP.	AUX. ENG. CODE	AUX. ENG. CODE	AUX. ENG. CODE	AUX. ENG. PEHF	AUX. ENG. OR THRUST	AUX. ENG. OR THRUST
CPROJ	CPINO	CPPAR	CPNUO	CGO	DELCD5	DELCD5	CXR	RTCLIM CODE	J	CP	CT	CT	CLW	CDW	RN
1.831	150.00	2251.4	15297.	1001.	69.5	1856.0	T	0.838	68.5	0.164	0.054	-3.1	11.6	1167.	1442.
725.0	978.	690.0	58.	0.01087	907.	0.00026	0.000296	A	0.055	0.055	0.055	0.055	0.007	0.949	0.949
0.000144	0.000161	0.000049	0.000007	0.01086	0.0	0.00026	0.000296	A	0.055	0.055	0.055	0.055	0.007	0.949	0.949
1.836	150.40	2246.6	15292.	1501.	53.5	1856.0	T	0.840	52.3	0.127	0.056	-2.1	16.4	1202.	1563.
725.0	1049.	690.0	60.	0.01033	894.	0.00007	0.000211	A	0.055	0.055	0.055	0.055	0.007	0.970	0.970
0.000133	0.000216	0.000027	0.000003	0.01033	0.0	0.00007	0.000211	A	0.055	0.055	0.055	0.055	0.007	0.970	0.970
1.842	150.68	2301.4	15287.	2001.	54.5	1856.0	T	0.841	52.9	0.129	0.057	-2.1	12.6	1220.	1230.
725.0	1027.	690.0	59.	0.01036	881.	0.00009	0.000217	A	0.055	0.055	0.055	0.055	0.007	0.969	0.969
0.000134	0.000226	0.000028	0.000003	0.01036	0.0	0.00009	0.000217	A	0.055	0.055	0.055	0.055	0.007	0.969	0.969
1.848	151.05	2307.4	15281.	2500.	55.5	1856.0	T	0.843	53.5	0.132	0.058	-2.1	12.0	1219.	1195.
725.0	1027.	690.0	59.	0.01039	870.	0.00011	0.000223	A	0.055	0.055	0.055	0.055	0.007	0.969	0.969
0.000135	0.000229	0.000029	0.000004	0.01039	0.0	0.00011	0.000223	A	0.055	0.055	0.055	0.055	0.007	0.969	0.969
1.855	151.43	2313.4	15275.	3001.	56.5	1856.0	T	0.844	54.1	0.134	0.059	-2.1	11.5	1219.	1159.
725.0	1027.	690.0	59.	0.01042	859.	0.00014	0.000229	A	0.055	0.055	0.055	0.055	0.007	0.968	0.968
0.000136	0.000233	0.000031	0.000004	0.01042	0.0	0.00014	0.000229	A	0.055	0.055	0.055	0.055	0.007	0.968	0.968

CRUISE AT SPEED FOR 99 PER CENT BEST RANGE WITH HEADWIND OF 0.0 KNOTS

TIME	RANGE	FUEL USED	WEIGHT	PRES. ALT.	TAS	PRIM. TURB. TEMP.	PRIM. ENG. CODE	PRIM. ENG. PEHF	EAS	MU	CT JVER	CT SIGMA	ALPHA D/L	SPEC. RANGE	BMP

7.6.2 Tandem Rotor Winged Helicopter

The design mission profile is illustrated in Figure 7-6. The more interesting and unusual inputs are discussed for this case, while the more routine information is only listed. A complete copy of the program printout follows the description of the input.

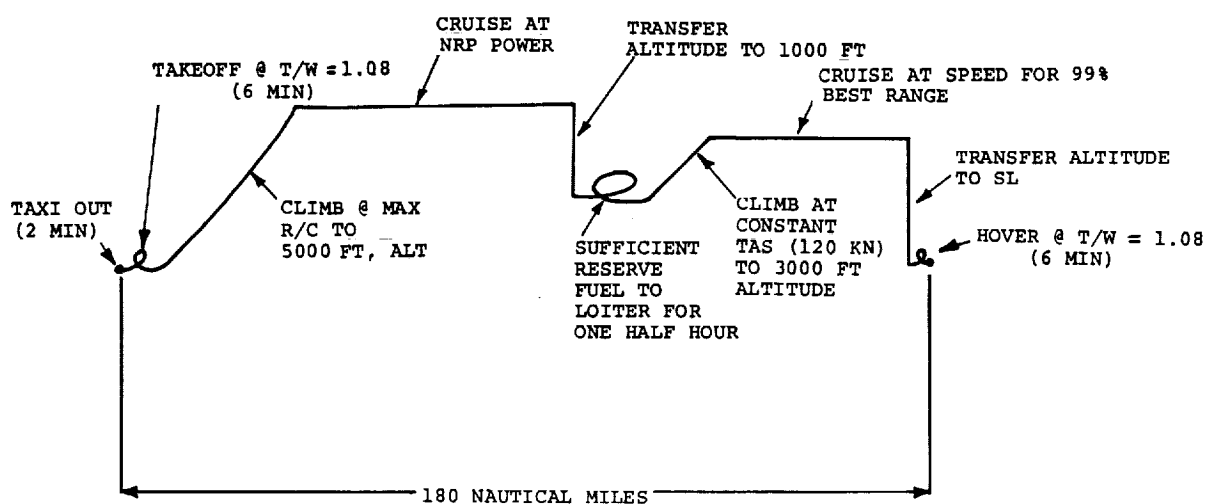


Figure 7-6. Design MSN - Sample Case Number 2

SAMPLE CASE NO. 2

GENERAL INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
OPTIND	0001	1.0	Sizing run
OPTIONAL PRINT	0002	1.0	Detailed printout de- sired
DRGIND	0003	1.0	Component drag build-up desired
OSWIND	0004	0	User inputs Oswald ef- ficiency factor
CNFIND	0005	2.0	Tandem rotor helicopter
AUXIND	0006	2.0	Winged helicopter
RDMIND	0007	4.0	Main rotor diameter sized based on input disc loading; solidity sized based on input C_T/σ
FIXIND	0008	1.0	Program sizes primary engines
ROTIND	0009	1.0	Short form rotor per- formance method used
S_W IND	0010	3.0	Wing area sized by ma- neuver conditions
b_W IND	0011	2.0	Wing span sized by in- put aspect ratio
AIPIND	0012	1.0	No independent aux. engines
FDMIND	0020	2.0	Tandem rotor fuselage sized by input of l_c and $((O/L)/D)$
APHIND	0021	2.0	Aft rotor pylon geom- etry calculated based on input rotor gap/ stagger ratio

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
ESCIND	0022	2.0	Primary engines sized for either takeoff or cruise
WG _O	0023	30000	First guess at design gross weight
h _O	0024	0	Start altitude
R _O	0025	0	Starting range
t _O	0026	0	Starting time
			} Normally 0 except for partial mission analysis
h _{OPT} ^{IND}	0027	0	Cruise at specified altitudes
M _{MO}	0028	0.32	Maximum operating Mach number
V _{MO}	0029	200	Maximum operating EAS knots
V _{DIVE}	0030	200	Design dive speed, knots EAS
M _{LF}	0031	3.0	Maneuver load factor
K ₁	0032	1.0	Factor on mission fuel burned to give reserve fuel; i.e., 1.1 would give 10 percent reserves
δ _{WF}	0033	0.0	Fixed fuel increment for reserves or other use
K _{FF}	0034	1.05	Increase basic engine SFC by 5 percent

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
SGTIND	0035	1.0	Taxi
	0036	2.0	Takeoff
	0037	3.0	Climb
	0038	4.0	Cruise
	0039	9.0	Transfer altitude
	0040	60.0	Loiter (reserve fuel)
	0041	3.0	Climb
	0042	4.0	Cruise
	0043	9.0	Transfer altitude
	0044	2.0	Hover and land
	0045	100	End of case
			Sequence of Design Mission

HELICOPTER DIMENSIONAL INFORMATION SHEET

AR	0104	6.0	Wing aspect ratio (input because $b_{WIND} = 2.0$)
$(t/c)_R$	0105	0.20	Wing root thickness/chord ratio
$(t/c)_T$	0106	0.12	Wing tip thickness/chord ratio
$\Lambda_c/4$	0107	0	Quarter-chord mean sweep angle, degrees
λ	0108	0.5	Wing taper ratio
C_F/C	0109	1.0	Ratio of download alleviating flap chord to wing chord (1.0 signifies a fully tilting wing)

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
h'/h_F	0110	1.0	Wing located at top of fuselage
C_{LD}	0111	1.2	Wing design lift coefficient
h_F	0122	7.0	Fuselage height
W_F	0123	6.5	Fuselage width
$(\ell/d)_P$	0124	0.70	Fineness ratio of nose
$(\ell/d)_T$	0125	1.2	Fineness ratio of tail
ℓ_C	0126	35.0	Constant diameter section length
ℓ_{RW}	0127	5.0	Length of ramp well
$((O/L)/D)$	0132	0.22	Tandem rotor overlap/diameter ratio
Z_1	0142	0.035	Primary engine nacelle constants
Z_2	0143	2.0	
Z_3	0144	0.078	
$(t/c)_{R_F}$	0152	0.45	Forward rotor pylon root thickness/chord ratio
$(t/c)_{T_F}$	0153	0.25	Forward rotor pylon tip thickness/chord ratio
AR_{FP}	0154	0.4	Forward rotor pylon aspect ratio
FP	0155	0.7	Forward rotor pylon taper ratio
h_{P_1}	0156	3.0	Forward rotor pylon height
$(t/c)_{R_A}$	0157	0.50	Aft rotor pylon root thickness/chord ratio
$(t/c)_{T_A}$	0158	0.30	Aft rotor pylon tip thickness/chord ratio

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
AR_{AP}	0159	0.7	Aft rotor pylon aspect ratio
λ_{AP}	0160	0.75	Aft rotor pylon taper ratio
g/s	0162	0.16	Tandem rotor gap/stagger ratio (input because $APHIND = 2.0$)

MAIN ROTOR DIMENSIONAL DATA SHEET

ROTOR CYCLE NO.	0171	3.0	Rotor blade section aerodynamic characteristics selection
N_R	0172	2.0	No. of rotors
W/A	0173	8.0	Disc loading
b_{MR}	0176	4.0	No. of blades/main rotor
θ_{TMR}	0177	-9.0	Main rotor twist (deg)
X_{CMR}	0178	0.2	Main rotor blade cutout as a fraction of radius
X_{MR}	0179	0.075	Main rotor blade attachment point as a fraction of radius
$(t/c)_{.25R}$	0180	0.12	Rotor blade thickness/chord at 25 percent radius
V_{TIP}	0181	700	Main rotor tip speed
$(C_T/\sigma)_H$	0182	0.12	Rotor "lift coefficient" for hover sizing rotor solidity
T/W	0183	1.08	Rotor design thrust/weight ratio

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
$V_{KT}(c)$	0184	160	Cruise flight conditions for sizing rotor solidity
$h_c(c)$	0185	4000	
ΔT_{INC}	0186	0	
$(C_T/\sigma)_{CR}$	0187	0.095	Rotor "lift coefficient" for sizing rotor solidity in cruise flight
g REQM'T	0188	2.0	Total g requirement helicopter must satisfy at $V_{KT}(c)$
g (ROTOR)	0189	1.5	Maneuver g's carried by main rotor at $V_{KT}(c)$
N (ROTOR LOADING)	0190	1.0	Rotor lift/GW for lg cruise flight rotor solidity sizing
V_{CEH1}	0191	1.53	Main rotor vertical rate-of-climb efficiency factors
V_{CEH2}	0192	0.0	
K_{PCLIMB}	0193	0.85	Helicopter forward flight climb efficiency

PRIMARY ENGINE SIZING INFORMATION SHEET

PRIMARY ENGINE	LOCATION	VALUE ASSIGNED	REMARKS
CYCLE NO.	0217	1.761	Engine selection
N_P	0219	4.0	No. of primary engines
XMSNIND	0220	2.0	Drive system rated at power required to hover or cruise (more critical of the two conditions selected by program)
SHP_{MRX}/SHP_{MR}^*	0221	1.00	Main rotor drive system is rated at 100 percent of main rotor design power
η_T	0223	0.97	Transmission efficiency

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
$\Delta \text{SHP}_{\text{ACC}}$	0224	100	Accessory power losses
$h_{\text{TO}}(\text{H})$	0227	4000	Design point hover altitude (engine sizing)
$(\text{T/W})_{\text{D}}$	0228	1.08	Configuration design point hover thrust/weight ratio
$(\text{N}_{\text{II}}/\text{N}_{\text{II MAX}})_{\text{TO}}$	0230	1.105	Main rotor operating at 100 percent of hover tip speed (700 fps); i.e.
			$\left(\frac{\text{N}_{\text{II}}}{\text{N}_{\text{II MAX}}} \right)_{\text{TO}} \left(\frac{\text{N}_{\text{II MAX}}}{\text{N}_{\text{II}^*}} \right) v_{\text{T}}$ $= \underbrace{(1.105)(.905)}_{1.00} (700)$ $= 700$
N_{PSD}	0231	1.0	One engine inoperative at hover design point conditions
$\text{SHP}_{\text{E}}/\text{SHP}^*$	0232	0.95	Engines sized to permit operation in hover (OGE) with one engine out and the remaining engines operating at 95 percent max rated power
h_{C}	0235	3000	Design point (cruise) altitude (engine sizing)
v_{C}	0236	155	Design point cruise speed (engine sizing)
CL_{DP}	0240	0.45	Wing operating lift coefficient at cruise condition for engine sizing
$(\text{N}_{\text{PSD}})_{\text{C}}$	0241	0.0	No. of primary engines shut down during cruise (for engine sizing)

HELICOPTER AERODYNAMICS INFORMATION SHEET

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
	0301		Use of the component drag build-up option (DRGIND = 1) is adequately explained in Section 4.9. For actual values used, refer to the appropriate locations on the output sheets.
	-0344		

ROTOR LIMITS INFORMATION SHEET

0347	This case has been run using dummy rotor limit values (as explained in Section 7.3). For actual dummy values used, refer to the appropriate locations on the output sheets.
-0377	

HELICOPTER WEIGHT INFORMATION SHEET

2601	This is fully explained in Section 4.11. Refer to the output sheets for the actual values used.
-2673	

TAXI INFORMATION

t_T	0411	0.0333	Taxi for 2 minutes
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TAKEOFF, HOVER, AND LANDING INFORMATION

TOLIND	0461	1.0	Specify required T/W for hover out of ground effect
	0462	1.0	
t_H	0551	0.1	Hover for 6 minutes
	0552	0.1	

CLIMB INFORMATION

CLMIND	0571	1.0	Climb at maximum rate of climb
	0572	4.0	Climb at constant TAS
C_{LWING}	0601	0.4	Wing operating C_L in climb
	0602	0.5	

VARIABLE	LOCATION	VALUE ASSIGNED	REMARKS
$N_{PSD_{CL}}$	0681	2.0	Shut down two of the primary engines in climb
	0682	2.0	
CRUISE INFORMATION			
CRSIND	0721	1.0	Cruise at specified power setting
	0722	4.0	Cruise at 99 percent best range speed
POWIND	0781	2.0	Cruise at NRP for first cruise segment
$C_{L_{WING}}$	0751	0.4	Wing operating C_L in cruise
	0752	0.45	
R_{MAX}	0791	80	Values of range at end of each cruise
	0792	180	
LOITER INFORMATION			
C_{L_W}	1041	0.4	Wing operating C_L in loiter
t_L	1081	0.5	Loiter 30 minutes for reserve fuel purposes
PRIMARY ENGINE CYCLE DATA; NON-STANDARD PERFORMANCE			
N_{2IND}	1204	2.0	A non-optimum N_{II} variation will be used
$N_{II_{MAX}}/N_{II}^*$	1223	0.905	Maximum power turbine speed is 90.5 percent of rated value (static, max power, sea level standard)

The sample case output follows:

H E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM B-91

THE FOLLOWING IS A CARD BY CARD REPRODUCTION OF THE INPUT DECK FOR THIS CASE

LOC. CORRESPONDS TO LOCATION NUMBER GIVEN ON INPUT SHEET
 NUM STANDS FOR THE NUMBER OF SEQUENTIAL INPUT VALUES STARTING WITH LOC. (MAX. =5)
 VAL EQUALS VALUE FOR VARIABLE CORRESPONDING TO LOC.
 VAL1 VALUE CORRESPONDING TO LOC.+0001
 VAL2 VALUE CORRESPONDING TO LOC.+0002
 ETC.

LOC.	NUM	VAL	VAL1	VAL2	VAL3	VAL4
------	-----	-----	------	------	------	------

NOTE : IN USING AUXILIARY ENGINES ; AUXILIARY ENGINE CYCLE INPUT LOCATIONS CAN BE CREATED BY PLACING A 66666 CARD IN FRONT AND BEHIND A STANDARD ENGINE CYCLE

1	1	1.0000				
2	1	1.0000				
3	1	1.0000				
4	1	0.0				
5	5	2.0000	2.0000	4.0000	1.0000	1.0000
10	3	3.0000	2.0000	1.0000		
20	5	2.0000	2.0000	2.0000	30000.	0.0
25	5	0.0	0.0	0.0	0.32000	200.00
30	5	200.00	3.0000	1.0000	0.0	1.0500
35	5	1.0000	2.0000	3.0000	4.0000	9.0000
40	5	60.000	3.0000	4.0000	9.0000	2.0000
45	1	100.00				
104	5	6.0000	0.20000	0.12000	0.0	0.50000
109	3	1.0000	1.0000	1.2000		
120	2	0.0	0.0			
122	5	7.0000	6.5000	0.70000	1.2000	35.000
127	1	5.0000				
132	1	0.22000				
142	4	0.35000E-01	2.0000	0.78000E-01	0.0	
152	5	0.45000	0.25000	0.40000	0.70000	3.0000
157	4	0.50000	0.30000	0.70000	0.75000	
162	1	0.16000				
171	1	3.0000				
172	2	2.0000	8.0000			
176	5	4.0000	-9.0000	0.20000	0.75000E-01	0.12000
181	1	700.00				
182	2	0.12000	1.0800			
184	5	160.00	4000.0	0.0	0.95000E-01	2.0000
189	2	1.5000	1.0000			
191	2	1.5300	0.0			
193	2	0.85000	0.0			
217	1	1.7610				
219	1	4.0000				
220	2	2.0000	1.0000			
223	2	0.97000	100.00			
227	5	4000.0	1.0800	50.300	1.1050	1.0000
232	2	0.95000	0.0			
234	5	2.0000	3000.0	155.00	0.0	1.1050
240	2	0.45000	0.0			

303	4	0.60000E-02	0.15000	0.75000	1.4000	
309	1	0.32000E-02				
314	1	0.75000				
316	1	5.5000				
319	3	1.0450	1.2500	1.3000		
323	1	2.8600				
326	2	1.2500	1.5400			
328	1	0.15440E 07				
329	1	6.2800				
330	1	6.0000				
331	5	0.0	0.20000	0.40000	0.60000	0.80000
336	1	1.0000				
339	5	0.59000E-02	0.60000E-02	0.68000E-02	0.80000E-02	0.95000E-02
344	1	0.11600E-01				
347	2	3.0000	3.0000			
349	3	0.0	0.50000	1.0000		
354	3	0.0	0.50000	1.0000		
361	3	1.0000	1.0000	1.0000		
368	3	1.0000	1.0000	1.0000		
375	3	1.0000	1.0000	1.0000		
2602	3	3000.0	600.00	5000.0		
2605	3	0.0	0.0	0.0		
2608	5	0.0	0.0	0.0	0.0	0.0
2613	5	26.000	18.000	42.000	0.0	0.0
2618	4	100.00	0.0	0.0	0.0	
2622	5	125.00	3.0000	0.40000E-01	0.80000	0.0
2627	5	1.0000	0.0	2.0600	0.0	0.0
2632	5	0.0	120.00	0.0	0.0	0.0
2637	5	44.000	1.0000	61.000	1.0000	1.2500
2642	5	0.0	0.0	0.0	0.0	250.00
2647	5	4.0000	0.0	0.0	0.0	0.19000
2652	2	200.00	0.0			
2654	5	1.0000	1.0000	1.0000	1.0000	1.0000
2659	5	1.0000	1.0000	1.0000	1.0000	1.0000
2664	1	1.0000				
2665	5	1.0000	1.0000	1.0000	1.0000	1.0000
2670	4	1.0000	1.0000	1.0000	1.0000	
401	1	0.0				
411	1	0.33300E-01				
421	1	0.0				
431	1	0.0				
441	1	1.1050				
461	2	1.0000	1.0000			
481	2	0.0	0.0			
501	2	0.0	0.0			
511	2	0.0	0.0			
521	2	1.0800	1.0800			
531	2	0.10000E-01	0.10000E-01			
541	2	1.1050	1.1050			
551	2	0.10000E 00	0.10000E 00			
571	2	1.0000	4.0000			
582	1	120.00				
591	2	0.0	0.0			
601	2	0.40000	0.50000			
611	2	0.0	0.0			
621	2	500.00	500.00			
631	2	2.0000	2.0000			
641	2	5000.0	3000.0			
651	2	1.1050	1.1050			
661	2	0.0	0.0			
671	2	1.1050	1.1050			

681	2	2.0000	2.0000			
701	2	0.0	0.0			
721	2	1.0000	4.1000			
732	1	25.0000				
741	2	0.0	0.0			
751	2	0.40000	0.45000			
761	2	0.0	0.0			
771	2	20.000	20.000			
781	2	2.0000	2.0000			
791	2	80.000	180.00			
801	2	1.1050	1.1050			
811	2	0.0	0.0			
821	2	1.1050	1.1050			
831	2	0.0	0.0			
851	2	0.0	0.0			
1031	1	0.0				
1041	1	0.40000				
1051	1	0.0				
1061	1	0.50000E-01				
1071	1	1.1050				
1081	1	0.50000				
1091	1	1.1050				
1101	1	0.0				
1121	1	0.0				
1131	1	0.0				
1181	2	1000.0	0.0			
1201	5	0.0	0.0	0.0	2.0000	0.0
1206	1	0.0				
1223	1	0.90500				
1201	5	0.0	0.0	0.0	2.0000	0.0
1206	1	0.0				
1223	1	0.90500				
1301	5	1.7610	0.15900	0.0	0.32000E-01	950.00
1306	5	1100.0	1856.0	2000.0	2000.0	8.0000
1311	5	950.00	1200.0	1400.0	1600.0	1800.0
1316	5	2000.0	2200.0	2600.0	5.0000	0.0
1321	4	0.20000	0.40000	0.60000	0.80000	
1326	5	0.25000E-01	0.25700E-01	0.27800E-01	0.31300E-01	0.36200E-01
1332	5	0.16300	0.16760	0.18130	0.20410	0.23600
1338	5	0.33500	0.34440	0.37250	0.41940	0.48510
1344	5	0.54400	0.55920	0.60490	0.68110	0.78770
1350	5	0.77000	0.79160	0.85620	0.96400	1.1150
1356	5	1.0000	1.0280	1.1120	1.2520	1.4480
1362	5	1.2000	1.2336	1.3344	1.5024	1.7376
1368	5	1.5500	1.5934	1.7236	1.9406	2.2444
1374	5	8.0000	950.00	1200.0	1400.0	1600.0
1379	5	1800.0	2000.0	2200.0	2600.0	5.0000
1384	5	0.0	0.20000	0.40000	0.60000	0.80000
1390	5	0.65000E-01	0.65100E-01	0.65300E-01	0.67000E-01	0.71000E-01
1394	5	0.11500	0.11600	0.11800	0.12800	0.14000
1402	5	0.18000	0.18100	0.19000	0.20800	0.22700
1408	5	0.26000	0.26100	0.27300	0.29500	0.32500
1414	5	0.34200	0.34700	0.36200	0.38900	0.42500
1420	5	0.42500	0.43500	0.45100	0.48600	0.51700
1426	5	0.50000	0.51100	0.53000	0.56000	0.61000
1432	5	0.62600	0.63100	0.66000	0.71800	0.78000
1438	4	3.0000	950.00	1600.0	2600.0	
1447	4	3.0000	0.0	0.40000	0.80000	
1454	3	0.26000	0.27100	0.29000		
1460	3	0.82000	0.84000	0.90000		
1466	3	1.0900	1.1180	1.1650		
1502	5	8.0000	950.00	1200.0	1400.0	1600.0
1507	5	1800.0	2000.0	2200.0	2600.0	5.0000
1512	5	0.0	0.20000	0.40000	0.60000	0.80000
1518	5	0.26000	0.26500	0.27100	0.28000	0.29000
1524	5	0.52000	0.52700	0.54000	0.56000	0.59000
1530	5	0.68000	0.69000	0.70500	0.73000	0.76000
1536	5	0.82000	0.82400	0.84000	0.86800	0.90000
1542	5	0.92000	0.93000	0.95000	0.98000	1.0200
1548	5	1.0000	1.0020	1.0200	1.0500	1.0900
1554	5	1.0520	1.0550	1.0700	1.1000	1.1310
1560	5	1.0900	1.1000	1.1180	1.1350	1.1650
1601	5	3.0000	-9.0000	0.99500E-02	-0.28000E-01	0.26200
1606	5	0.27600	2.4500	0.86500	0.10500E-01	2.8200
1611	5	0.90000E-01	1.1700	0.12400E-02	0.75800	0.74300
1616	5	10.000	0.0	0.40000E-02	0.70000E-02	0.90000E-02
1621	5	0.10000E-01	0.11000E-01	0.11500E-01	0.12000E-01	0.15500E-01
1626	5	0.22000	1.0180	1.0850	1.1540	1.2330
1631	5	1.2790	1.3140	1.3270	1.3370	1.3640
1636	1	1.3970				

H E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM 8-91

TANDEM ROTOR WINGED HELICOPTER

S I Z E C A T A THIS RUN CONVERGED IN 3 ITERATIONS

GROSS WEIGHT = 24482. LB

FUSELAGE

LF	LENGTH	47.8 FT.
LC	CABIN LENGTH	35.0 FT.
DELTA1	FWD. ROTOR LOCATION	9.2 FT.
DELTA2	AFT ROTOR LOCATION	4.6 FT.
WF	WIDTH	6.5 FT.
G/S	ROTOR GAP/STAGGER RATIO	0.160
(C/L/D)	ROTOR OVERLAP/DIAMETER RATIO	0.220
SF	WETTED AREA	936.4 SQ. FT.

WING

AR	ASPECT RATIO	6.00
SW	AREA	132.3 SQ. FT.
PN	SPAN	28.2 FT.
CBARW	MEAN CHORD	4.7 FT.
LAMBDA C/4	QUARTER CHORD SWEEP	0.0 DEG.
LAMBDA	TAPER RATIO	0.500
(T/C)R	ROOT THICKNESS/CHORD	0.200
(T/C)T	TIP THICKNESS/CHORD	0.120
WC/SW	WING LOADING	185.0 LBS/SQ. FT.
GRW	ROTOR/WING GAP	3.0 FT.
CF/C	FLAP CHORD/MEAN CHORD RATIO	1.000

FORWARD ROTOR PYLON

AR	ASPECT RATIO	0.400
SFP	WETTED AREA	50.0 SQ. FT.
FAPF	FRONTAL AREA	8.3 SQ. FT.
HPI	HEIGHT	3.0 FT.
CBARFP	MEAN CHORD	7.5 FT.
LAMBDA FP	TAPER RATIO	0.700
(T/C)R	ROOT THICKNESS/CHORD	0.450
(T/C)T	TIP THICKNESS/CHORD	0.250

AFT ROTOR PYLON

AR	ASPECT RATIO	0.700
SAP	WETTED AREA	232.0 SQ. FT.
HP2	HEIGHT	8.4 FT.
CBARAP	MEAN CHORD	12.1 FT.
LAMBDA AP	TAPER RATIO	0.750
(T/C)R	ROOT THICKNESS/CHORD	0.500
(T/C)T	TIP THICKNESS/CHORD	0.300

PRIMARY ENGINE NACELLE

LA	LENGTH	5.2 FT.
CA	MEAN DIAMETER	1. FT.
SN	WETTED AREA (TOTAL FOR ALL ENGINES)	92.6 SQ. FT.

AUXILIARY INDEPENDENT ENGINE NACELLE -NO AUXILIARY INDEPENDENT ENGINE USED

PROPELLER(AUXILIARY PROPULSION) - NO PROPELLER USED

MAIN ROTOR

EMP	DIAMETER	44.1 FT.
SICMR	SOLIDITY	0.122
WG/A	DISC LOADING	8.0 LB/SQ. FT.
CT/SIGMA	THRUST COEFF./SOLIDITY	0.095
NR	NO. OF ROTORS	2.
NO. BLADES	NO. OF BLADES/ROTOR	4.
THETA	BLADE TWIST	-9.000 DEG.
XC	BLADE CUTOFF/RADIUS RATIO	0.200
VTIP	TIP SPEED	700. FT./SEC.

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WEIGHTS DATA IN LBS

MLF	MANEUVER LOAD FACTOR	3.000	
CLF	GUST LOAD FACTOR	1.438	
LLF	ULTIMATE LOAD FACTOR	4.500	
PROPULSION GROUP			
WPRG	TOTAL MAIN ROTOR GROUP	2893.	
K12 WPRB	MAIN ROTOR BLADE (PER ROTOR)	587.	
K13 WPH	MAIN ROTOR HUB (PER ROTOR)	570.	
WRF	BLADE FOLDING (PER ROTOR)	289.	
K15 WAR	AUXILIARY PROPULSION ROTOR GROUP	0.	
WDS	DRIVE SYSTEM	2230.	
K16 WPOS	MAIN ROTOR DRIVE SYSTEM	2230.	
K20 WTRDS	TAIL ROTOR DRIVE SYSTEM	0.	
K17 WADS	AUXILIARY PROPULSION DRIVE SYSTEM	0.	
K18 WEP	PRIMARY ENGINES	1053.	
K19 WEA	AUXILIARY ENGINES	0.	
WPEI	PRIMARY ENGINE INSTALLATION	200.	
WAEI	AUXILIARY ENGINE INSTALLATION	0.	
WFS	FUEL SYSTEM	618.	
DELTA WP	PROPULSION GROUP WEIGHT INCREMENT	0.	
WP	TOTAL PROPULSION GROUP WEIGHT		6993.
STRUCTURES GROUP			
K8 WW	WING	273.	
WTG	TAIL GROUP	0.	
K9 WMT	HOR. TAIL	0.	
K14 WTR	TAIL ROTOR	0.	
K6 WB	FUSELAGE	3229.	
K7 WLG	LANDING GEAR	979.	
WNG	NOSE GEAR	196.	
WMG	MAIN GEAR	783.	
WTES	TOTAL ENGINE SECTION	120.	
WPES	PRIMARY ENGINE SECTION	120.	
WAES	AUXILIARY ENGINE SECTION	0.	
DELTA WST	STRUCTURE WEIGHT INCREMENT	0.	
WST	TOTAL STRUCTURE WEIGHT		4601.
FLIGHT CONTROLS GROUP			
WPFC	PRIMARY FLIGHT CONTROLS	1036.	
WCC	COCKPIT CONTROLS	96.	
K1 WRC	MAIN ROTOR CONTROLS	252.	
K2 WSC	MAIN ROTOR SYSTEMS CONTROLS	588.	
K3 WFW	FIXED WING CONTROLS	0.	
WTM	TILT MECHANISM	0.	
WSAS	SAS	100.	
WAFS	AUXILIARY FLIGHT CONTROLS	0.	
K4 WRCA	AUX. PROPULSION ROTOR CONTROLS	0.	
K5 WSCA	AUX. PROPULSION ROTOR SYS. CONTROLS	0.	
MISCELLANEOUS CONTROLS			
WMC	MISCELLANEOUS CONTROLS	0.	
DELTA WFC	CONTROL WEIGHT INCREMENT	0.	
WFC	TOTAL CONTROL WEIGHT		1036.
WFE	WEIGHT OF FIXED EQUIPMENT		3000.
WE	WEIGHT EMPTY		15630.
WFUL	FIXED USEFUL LOAD		600.
CWE	OPERATING WEIGHT EMPTY		16230.
WPL	PAYLOAD		5000.
(WFA)	FUEL		3252.
WG	GROSS WEIGHT		24482.

SAMPLE CASE #2

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R O T O R D A T A

ROTOR CYCLE NO. 3.0000

MAIN ROTOR SCLIDITY SIZED BY MANUEVER CONDITIONS
H = 4000.0 FT. , TEMP = 44.7 DEG. , V = 160.0 KT.
ROTOR MANUEVER G'S = 1.500 , CT/SIGMA = 0.095

SAMPLE CASE #2

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H E S C O M P
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P R O P U L S I O N D A T A
PRIMARY PROPULSION CYCLE NO. 1.761
TURBOSHAFT ENGINE

4. ENGINES

BHPBP MAX. STANDARD S.L. STATIC H.P. 6621. H.P.

ENGINE SIZED FOR TAKEOFF AT T/M = 1.08
H = 4000. FT, TEMPERATURE = 95.04 DEG.F.
AND 1.000 ENGINES INOPERATIVE.

MAIN ROTOR DRIVE SYSTEM RATING 3597. H.P.

XMSN SIZED AT 100. PERCENT OF MAIN ROTOR HOVER POWER REQUIRED
AT H = 4000. FT, TEMP = 95.04 DEG.F.

SAMPLE CASE #2

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H E S C O M P
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A E R O D Y N A M I C S C A T A

FF	TOTAL EFFECTIVE FLATPLATE AREA	23.477	SQFT
SWET	TOTAL WETTED AREA	1521.	SQFT
CPARF	MEAN SKIN FRICTION COEFF.	0.015434	
D R A G A R E A K O C W N IN SQFT			
FEW	WING FE	1.267	
FEF	FUSELAGE FE	2.479	
FEFP	FORWARD(MAIN) ROTOR PYLON FE	1.551	
FEAP	AFT ROTOR PYLON FE	1.318	
FEFRH	MAIN ROTOR FUP(S) FE	11.925	
FFTRH	TAIL ROTOR FUP FE	0.0	
FEVT	VERTICAL TAIL FE	0.0	
FEHT	HORIZONTAL TAIL FE	0.0	
FEA	PRIMARY ENGINE NACELLE FE	0.879	
FEAI	AUX. INDEPENDENT CRUISE ENG. NAC. FE	0.0	
FEAS	AUX. INDEPENDENT CRUISE ENG. STRUT FE	0.0	
CEFLTA FE	INCREMENTAL FE	4.058	

A E R O D Y N A M I C C O E F F .

A5		22.70988
A6		1.62279
A7		0.07074
A8		0.00010
A9		0.0
E	WING LIFT EFFICIENCY FACTOR	0.75000
EVT	VERTICAL TAIL LIFT EFFICIENCY FACTOR	0.0

P E S C O M P
HELICOPTER SIZING & PERFORMANCE COMPUTER PROGRAM 8-91

MISSION PERFORMANCE DATA

TAXI FOR 0.033 HRS. AT GROUND IDLE ENGINE RATING

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRESS. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PEHF	TOTAL FUEL FLOW (LBS/HR)	AUX. TURB. TEMP. (R)	AUX. ENG. CODE	AUX. ENG. PEHF	AUX. ENG. FUEL FLOW (LBS/HR)
0.0	0.0	0.0	24482.	0.	0.0	950.0	T	0.0	452.	*****	****	*****	0.
0.033	0.0	15.0	24467.	0.	0.0	950.0	T	0.0	452.	*****	****	*****	0.

TAKEOFF, HOVER, OR LAND AT T/M = 1.080 FOR 0.100 HRS.

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRESS. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PEHF	TOTAL FUEL FLOW (LBS/HR)	THRUST TO WEIGHT	FM	BHP	CT	CT/SIGMA
M. PCTOR VTIP (FPS)	M. PCTOR RHP	T. PCTOR VTIP (FPS)	T. PCTOR RHP	VPC RHP	PRIM. ENG. FUEL FLOW (LBS/HR)	AUX. ENG. FUEL FLOW (LBS/HR)	ROT LIM CODE			DELDCM	FMI	CPROD	CPIND	CDO
0.033 700.0	0.0 3034.	15.0 *****	24467. 0.	0. 0.	0.0 1740.	1574.0 0.	P A	0.488	1740.	1.080 0.0	0.675 0.708	3228. 0.00011	0.0074 0.00056	0.061 0.0092
0.043 700.0	0.0 3031.	32.4 *****	24450. 0.	0. 0.	0.0 1739.	1575.6 0.	P A	0.487	1739.	1.080 0.0	0.675 0.708	3225. 0.00011	0.0074 0.00056	0.061 0.0092
0.053 700.0	0.0 3028.	49.8 *****	24432. 0.	0. 0.	0.0 1738.	1575.2 0.	P A	0.487	1738.	1.080 0.0	0.675 0.708	3222. 0.00011	0.0074 0.00055	0.061 0.0092
0.063 700.0	0.0 3025.	67.2 *****	24415. 0.	0. 0.	0.0 1737.	1574.9 0.	P A	0.486	1737.	1.080 0.0	0.675 0.708	3218. 0.00011	0.0074 0.00055	0.061 0.0092
0.073 700.0	0.0 3022.	84.6 *****	24398. 0.	0. 0.	0.0 1736.	1574.5 0.	P A	0.486	1736.	1.080 0.0	0.675 0.708	3215. 0.00011	0.0074 0.00055	0.061 0.0092
0.083 700.0	0.0 3019.	101.9 *****	24380. 0.	0. 0.	0.0 1735.	1574.1 0.	P A	0.485	1735.	1.080 0.0	0.675 0.708	3212. 0.00011	0.0074 0.00055	0.060 0.0092
0.093 700.0	0.0 3015.	119.3 *****	24363. 0.	0. 0.	0.0 1734.	1573.7 0.	P A	0.485	1734.	1.080 0.0	0.675 0.708	3209. 0.00011	0.0074 0.00055	0.060 0.0092
0.103 700.0	0.0 3012.	136.6 *****	24346. 0.	0. 0.	0.0 1733.	1573.3 0.	P A	0.484	1733.	1.080 0.0	0.675 0.708	3205. 0.00011	0.0074 0.00055	0.060 0.0092
0.113 700.0	0.0 3009.	154.0 *****	24328. 0.	0. 0.	0.0 1731.	1572.9 0.	P A	0.484	1731.	1.080 0.0	0.675 0.708	3202. 0.00011	0.0074 0.00055	0.060 0.0092
0.123 700.0	0.0 3006.	171.3 *****	24311. 0.	0. 0.	0.0 1730.	1572.6 0.	P A	0.483	1730.	1.080 0.0	0.675 0.708	3199. 0.00011	0.0074 0.00055	0.060 0.0092

CLIMB TO 5000. FT. WITH MAXIMUM R/C AT NORMAL ENGINE RATING

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. PENF	EAS (KTS)	MU	CT PRIME OVER SIGMA	ALPHA D/L (DEG)	GAMMA (DEG)	BHP (FPM)	R/C (FPM)
M. ROTOR RHP (FPS)	M. ROTOR RHP	T. ROTOR VTIP (FPS)	T. ROTOR RHP	PROP VTIP (FPS)	PRIM. ENG FUEL FLOW (LBS/HR)	BHP AUX (LBS/HR)	ETAP PROP	TAXI/T FUEL FLOW (LBS/HR)	J	CP	CT	CLM	CDM	RN	
CPPRO	CPIND	CPPAR	CPNUD	CDO	DELCDL	DELCDL	CXR	ROT LIM CODE							
0.123 700.0 0.000155	0.0 1828. 0.000187	171.3 0.000049	24311. 0.000049	0. 0.01067	83.4 1288. 0.0	1856.0 0.00021	T 0.000237	0.835 A	83.4 0.0000	0.203 0.0000	0.053 0.0000	-2.1 0.400	6.2 0.007	1985. 0.947	924.
0.132 700.0 0.000145	0.76 1881. 0.000236	182.9 0.000032	24299. 0.000007	500. 0.01050	69.4 1268. 0.0	1856.0 0.00008	T 0.000187	0.837 A	68.9 0.0000	0.170 0.0000	0.054 0.0000	-1.6 0.400	7.8 0.007	2039. 0.964	971.
0.141 700.0 0.000146	1.36 1887. 0.000241	193.8 0.000033	24289. 0.000008	1000. 0.01052	70.4 1250. 0.0	1856.0 0.00009	T 0.000191	0.838 A	69.4 0.0000	0.172 0.0000	0.055 0.0000	-1.6 0.400	5.9 0.007	2046. 0.963	744.
0.152 700.0 0.000146	2.15 1894. 0.000249	207.8 0.000033	24275. 0.000008	1500. 0.01052	70.4 1235. 0.0	1856.0 0.00010	T 0.000192	0.840 A	68.8 0.0000	0.172 0.0000	0.056 0.0000	-1.6 0.400	5.7 0.007	2053. 0.964	713.
0.164 700.0 0.000148	2.99 1886. 0.000249	222.2 0.000036	24260. 0.000008	2000. 0.01056	72.4 1220. 0.0	1856.0 0.00014	T 0.000200	0.841 A	70.3 0.0000	0.177 0.0000	0.057 0.0000	-1.7 0.400	5.2 0.007	2045. 0.962	672.
0.176 700.0 0.000149	3.89 1886. 0.000253	237.3 0.000037	24245. 0.000009	2500. 0.01059	73.4 1205. 0.0	1856.0 0.00016	T 0.000205	0.843 A	70.7 0.0000	0.179 0.0000	0.058 0.0000	-1.7 0.400	4.9 0.007	2044. 0.962	651.
0.189 700.0 0.000150	4.84 1886. 0.000257	252.8 0.000039	24230. 0.000009	3000. 0.01062	74.4 1190. 0.0	1856.0 0.00019	T 0.000209	0.844 A	71.2 0.0000	0.182 0.0000	0.059 0.0000	-1.7 0.400	4.6 0.007	2044. 0.961	619.
0.202 700.0 0.000151	5.86 1886. 0.000261	268.8 0.000040	24213. 0.000010	3500. 0.01066	75.4 1175. 0.0	1856.0 0.00023	T 0.000214	0.846 A	71.6 0.0000	0.184 0.0000	0.059 0.0000	-1.7 0.400	4.3 0.007	2045. 0.961	587.
0.217 700.0 0.000152	6.94 1887. 0.000265	285.5 0.000041	24197. 0.000010	4000. 0.01070	76.4 1160. 0.0	1856.0 0.00027	T 0.000219	0.847 A	72.0 0.0000	0.187 0.0000	0.060 0.0000	-1.7 0.400	4.0 0.007	2046. 0.961	554.
0.232 700.0 0.000153	8.10 1888. 0.000269	302.9 0.000043	24179. 0.000011	4500. 0.01075	77.4 1145. 0.0	1856.0 0.00031	T 0.000223	0.848 A	72.4 0.0000	0.189 0.0000	0.061 0.0000	-1.7 0.400	3.8 0.007	2047. 0.960	521.
0.248 700.0	9.36 1890.	321.3 0.000044	24161. 0.000011	5000. 0.01080	78.4 1131. 0.0	1856.0 0.00035	T 0.000227	0.850 A	72.8 0.0000	0.191 0.0000	0.062 0.0000	-1.7 0.400	3.5 0.007	2049. 0.959	487.

0.000154 0.000274 0.000044 0.000011 0.01080 0.00000 0.00036 0.000228 A ***** 0.400 0.007 0.960

CRUISE AT ACPAL ENGINE RATING

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WGT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PERM	EAS (KTS)	MU	CT PRIME OVER SIGMA	ALPHA D/L (DEG)	SPEC. RANGE (NMPS)	BHP
M.ROTOR VTIP (FPS)	M.ROTOR RHP	T.ROTOR VTIP (FPS)	T.ROTOR RHP	PROP VTIP (FPS)	PRIM. ENG. FUEL FLOW (LBS/HRS)	BHP AUX	ETAP PROP	AUX. ENG. TAUX/T FUEL FLOW (LBS/HRS)	AUX. TURB. TEMP.	AUX. ENG. CODE	AUX. ENG. PERM	AUX. ENG. PERM	AUX. ENG. BHP CP THRUST	

CPPRO	CPIND	CPPAR	CPNUD	CDU	CELCCS	DELCDM	CXR	ROTLIM CODE	J	CP	CT	CLW	CDW	RN
0.248	9.36	321.3	24161	5000	186.0	1856.0	T	0.851	172.7	0.449	0.050	-8.3	-08049	5255
700.0	5000	0.000078	0.000031	0.02384	0.00083	0.01252	0.000890	A	0.0	0.0	0.0	0.400	0.007	0.779
0.000563	0.000076	0.000609	0.000030	0.02378	0.00078	0.01250	0.000892	A	0.0	0.0	0.0	-8.4	-08061	5254
700.0	4998	0.000074	0.000030	0.02371	0.00074	0.01248	0.000894	A	0.0	0.0	0.0	0.400	0.007	0.776
0.000562	0.000072	0.000616	0.000030	0.02365	0.00069	0.01246	0.000895	A	0.0	0.0	0.0	-8.5	-08072	5253
700.0	4997	0.000071	0.000029	0.02362	0.00067	0.01245	0.000896	A	0.0	0.0	0.0	0.400	0.007	0.773
0.570	69.36	1065.6	23417	5000	186.8	1856.0	T	0.851	173.4	0.451	0.048	-8.7	-08083	5252
700.0	4998	0.000072	0.000030	0.02365	0.00069	0.01246	0.000895	A	0.0	0.0	0.0	0.400	0.007	0.769
0.000561	0.000070	0.000615	0.000029	0.02362	0.00067	0.01245	0.000896	A	0.0	0.0	0.0	-8.7	-08089	5252
700.0	4997	0.000071	0.000029	0.02362	0.00067	0.01245	0.000896	A	0.0	0.0	0.0	0.400	0.007	0.768

TRANSFER ALTITUDE TO 1000. FT.

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WGT (LBS.)	PRES. ALT. (FT)
0.627	80.00	1197.3	23285	5000
0.627	80.00	1197.3	23285	1000

LOITER FOR 0.500 MRS. FOR RESERVE FUEL

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WGT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PERM	EAS (KTS)	MU	CT PRIME OVER SIGMA	ALPHA D/L (DEG)	TOTAL FUEL FLOW (LBS/HRS)	BHP
M.ROTOR VTIP (FPS)	M.ROTOR RHP	T.ROTOR VTIP (FPS)	T.ROTOR RHP	PROP VTIP (FPS)	PRIM. ENG. FUEL FLOW (LBS/HRS)	BHP AUX	ETAP PROP	AUX. ENG. TAUX/T FUEL FLOW (LBS/HRS)	AUX. TURB. TEMP.	AUX. ENG. CODE	AUX. ENG. PERM	AUX. ENG. PERM	AUX. ENG. BHP CP THRUST	

CPPRO	CPIND	CFPAR	CPNUD	COO	DELCDS	DELCDM	CXR	ROTLIM CODE	J	CP	CT	CLW	CDW	RN
0.627	80.00	1828.9	23285.	1000.	120.0	1856.0	T	0.838	118.3	0.289	0.048	-4.3	2.8	2256.599.
700.0	2092.	0.000109	0.000027	0.01193	1262.	0.00143	0.000433	A	0.	0.0000	0.0000	0.500	0.007	0.865
0.000202	0.000113	0.000137	0.000028	0.01201	0.00000	0.00151	0.000434	A	0.	0.0000	0.0000	0.500	0.007	0.867
0.641	81.67	1846.4	23267.	1500.	120.0	1856.0	T	0.840	117.4	0.289	0.048	-4.2	2.7	2251.571.
700.0	2086.	0.000113	0.000028	0.01201	1246.	0.00151	0.000434	A	0.	0.0000	0.0000	0.500	0.007	0.867
0.000204	0.000113	0.000137	0.000028	0.01201	0.00000	0.00151	0.000434	A	0.	0.0000	0.0000	0.500	0.007	0.867
0.655	83.42	1864.6	23249.	2000.	120.0	1856.0	T	0.842	116.5	0.289	0.049	-4.1	2.6	2245.542.
700.0	2081.	0.000116	0.000137	0.000028	1231.	0.00159	0.000435	A	0.	0.0000	0.0000	0.500	0.007	0.869
0.000205	0.000116	0.000137	0.000028	0.01209	0.00000	0.00159	0.000435	A	0.	0.0000	0.0000	0.500	0.007	0.869
0.671	85.27	1883.6	23230.	2500.	120.0	1856.0	T	0.843	115.7	0.289	0.050	-4.1	2.4	2241.513.
700.0	2076.	0.000120	0.000138	0.000028	1216.	0.00168	0.000436	A	0.	0.0000	0.0000	0.500	0.007	0.871
0.000207	0.000120	0.000138	0.000028	0.01218	0.00001	0.00168	0.000436	A	0.	0.0000	0.0000	0.500	0.007	0.871
0.687	87.22	1903.3	23210.	3000.	120.0	1856.0	T	0.844	114.8	0.289	0.051	-4.0	2.3	2236.483.
700.0	2072.	0.000124	0.000138	0.000029	1201.	0.00177	0.000438	A	0.	0.0000	0.0000	0.500	0.007	0.873
0.000208	0.000124	0.000138	0.000029	0.01227	0.00001	0.00177	0.000438	A	0.	0.0000	0.0000	0.500	0.007	0.873

CRUISE AT SPEED FOR 99 PER CENT BEST RANGE WITH HEADWIND OF 25.0 KNOTS

TIME (HRS)	RANGE (N.M.)	FUEL USED (LBS)	WEIGHT (LBS.)	PRES. ALT. (FT)	TAS (KTS)	PRIM. TURB. TEMP. (R)	PRIM. ENG. CODE	PRIM. ENG. PEHF	EAS (KTS)	MU	CT PRIME OVER SIGMA	ALPHA D/L (DEG)	SPEC. RANGE (NMPP)	BHP
0.687	87.22	1503.3	23210.	3000.	159.6	1611.2	P	0.550	152.6	0.385	0.046	-6.9	0.07742	3512.
700.0	3309.	0.000078	0.000338	0.01647	1740.	0.00586	0.000686	A	0.	0.0000	0.0000	0.450	0.007	0.797
0.000340	0.000078	0.000338	0.000041	0.01647	0.00011	0.00586	0.000686	A	0.	0.0000	0.0000	0.450	0.007	0.797
0.836	107.22	2161.7	22952.	3000.	159.6	1608.8	P	0.547	152.6	0.385	0.046	-7.0	0.07772	3492.
700.0	3291.	0.000076	0.000337	0.01641	1733.	0.00581	0.000685	A	0.	0.0000	0.0000	0.450	0.007	0.795
0.000338	0.000076	0.000337	0.000041	0.01641	0.00010	0.00581	0.000685	A	0.	0.0000	0.0000	0.450	0.007	0.795
0.984	127.22	2419.0	22695.	3000.	159.6	1606.4	P	0.544	152.6	0.385	0.045	-7.1	0.07801	3474.
700.0	3272.	0.000074	0.000337	0.01635	1727.	0.00576	0.000684	A	0.	0.0000	0.0000	0.450	0.007	0.793
0.000337	0.000074	0.000337	0.000040	0.01635	0.00009	0.00576	0.000684	A	0.	0.0000	0.0000	0.450	0.007	0.793
1.133	147.22	2875.4	22438.	3000.	159.1	1601.1	P	0.538	152.2	0.384	0.045	-7.1	0.07837	3431.
700.0	3231.	0.000072	0.000333	0.01622	1712.	0.00564	0.000679	A	0.	0.0000	0.0000	0.450	0.007	0.792
0.000334	0.000072	0.000333	0.000040	0.01622	0.00008	0.00564	0.000679	A	0.	0.0000	0.0000	0.450	0.007	0.792
1.282	167.22	2530.6	22183.	3000.	159.1	1598.8	P	0.535	152.2	0.384	0.044	-7.2	0.07866	3413.

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